20000038013

NASA
Center for AeroSpace Information (CASI)
Global Connection to Aerospace Research

**DISTRIBUTION STATEMENT A**
Approved for Public Release
Distribution Unlimited

***A Service of:***
National Aeronautics and
Space Administration

STI
SCIENTIFIC & TECHNICAL INFORMATION

20000821 018

RTO-TR-28
AC/323(AVT)TP/22

RTO-TR-28

NORTH ATLANTIC TREATY ORGANIZATION

RESEARCH AND TECHNOLOGY ORGANIZATION

BP 25, 7 RUE ANCELLE, F-92201 NEUILLY-SUR-SEINE CEDEX, FRANCE



**RTO TECHNICAL REPORT 28**

# Recommended Practices for Monitoring Gas Turbine Engine Life Consumption

(Pratiques recommandées pour le contrôle du vieillissement des turbomoteurs)

*Report of the Applied Vehicle Technology Panel Task Group AVT-017.*

Published April 2000

# The Research and Technology Organization (RTO) of NATO

RTO is the single focus in NATO for Defence Research and Technology activities. Its mission is to conduct and promote cooperative research and information exchange. The objective is to support the development and effective use of national defence research and technology and to meet the military needs of the Alliance, to maintain a technological lead, and to provide advice to NATO and national decision makers. The RTO performs its mission with the support of an extensive network of national experts. It also ensures effective coordination with other NATO bodies involved in R&T activities.

RTO reports both to the Military Committee of NATO and to the Conference of National Armament Directors. It comprises a Research and Technology Board (RTB) as the highest level of national representation and the Research and Technology Agency (RTA), a dedicated staff with its headquarters in Neuilly, near Paris, France. In order to facilitate contacts with the military users and other NATO activities, a small part of the RTA staff is located in NATO Headquarters in Brussels. The Brussels staff also coordinates RTO's cooperation with nations in Middle and Eastern Europe, to which RTO attaches particular importance especially as working together in the field of research is one of the more promising areas of initial cooperation.

The total spectrum of R&T activities is covered by 7 Panels, dealing with:

- SAS Studies, Analysis and Simulation
- SCI Systems Concepts and Integration
- SET Sensors and Electronics Technology
- IST Information Systems Technology
- AVT Applied Vehicle Technology
- HFM Human Factors and Medicine
- MSG Modelling and Simulation

These Panels are made up of national representatives as well as generally recognised 'world class' scientists. The Panels also provide a communication link to military users and other NATO bodies. RTO's scientific and technological work is carried out by Technical Teams, created for specific activities and with a specific duration. Such Technical Teams can organise workshops, symposia, field trials, lecture series and training courses. An important function of these Technical Teams is to ensure the continuity of the expert networks.

RTO builds upon earlier cooperation in defence research and technology as set-up under the Advisory Group for Aerospace Research and Development (AGARD) and the Defence Research Group (DRG). AGARD and the DRG share common roots in that they were both established at the initiative of Dr Theodore von Kármán, a leading aerospace scientist, who early on recognised the importance of scientific support for the Allied Armed Forces. RTO is capitalising on these common roots in order to provide the Alliance and the NATO nations with a strong scientific and technological basis that will guarantee a solid base for the future.

The content of this publication has been reproduced directly from material supplied by RTO or the authors.

Published April 2000



ISBN 92-837-1032-0

*Printed by Canada Communication Group Inc.*
*(A St. Joseph Corporation Company)*
*45 Sacré-Cœur Blvd., Hull (Québec), Canada K1A 0S7*

# Recommended Practices for Monitoring Gas Turbine Engine Life Consumption

## (RTO TR-28)

## Executive Summary

The main reason for usage monitoring is to ensure that gas-turbine engine components do not fail in service. Early maintenance policies were based on a "hard-time" philosophy, measured in hours of use. More recently, technical advances have permitted a trend towards a condition-based philosophy. Life monitoring systems have evolved to automate measurement of the life used in modern engines. In many cases, the new assessment and monitoring methods have been applied to ageing fleets with great success. This document describes the design and operational factors that should be considered prior to the application of these techniques to an aged engine fleet.

Turbine disks are the most safety critical parts, due to the amount of energy released should they fail. Moreover, future improvements in engine performance directly depend on increases in component stress levels. These demands for better performance must be met within the regulatory requirements for safety standards. This situation presents one of the most challenging areas of gas turbine design.

Satisfying these conflicting demands carries a cost that reaches far beyond the development and purchase costs of a particular engine design. Although turbine disks are particularly expensive to make, over 90% of them are thrown away when less than 50% of their life has been consumed. Physical use, component life and safety have to be managed on a statistical basis because of variations in material properties.

Because of the overwhelming prevalence of mature fleets, consideration was restricted to aircraft that entered service prior to 1990. This keeps it in the realm of feasible and relevant solutions for operators with ageing fleets. The document remains relevant for new projects begun after 1990. The aim has been:

- To cover technical and managerial issues in a manner that allows readers to concentrate on those parts that interest them. This has necessitated some repetition between chapters;
- To show the parallel regulatory developments in the USA and Europe;
- To make clear the difference between 'safe life' and 'damage tolerance' design philosophies;
- To show the impact of differing approaches to maintenance practice;
- To show the results obtained from existing monitoring systems;
- To give practical help and guidance to operators, with mature fleets.

Within the forum, the leading design and life management practitioners from the world's major engine design companies worked with representatives from most of the NATO aircraft operators.

# Pratiques recommandées pour le contrôle du vieillissement des turbomoteurs

## (RTO TR-28)

# Synthèse


L'objectif principal du contrôle du vieillissement est d'éviter la défaillance en vol des composants des turbopropulseurs. Les premières politiques de maintenance suivaient une philosophie de « cycles », mesurés en heures d'exploitation. Plus récemment, les avancées technologiques réalisées ont permis l'adoption d'une politique basée sur les conditions d'utilisation. L'évolution des systèmes de contrôle du vieillissement a été telle qu'ils sont désormais en mesure de calculer automatiquement le nombre d'heures de fonctionnement restant pour les moteurs modernes. Dans de nombreux cas, ces nouvelles méthodes d'évaluation et de contrôle du vieillissement ont été appliquées avec beaucoup de succès à des flottes aériennes vieillissantes. Ce document décrit les critères de conception et les facteurs opérationnels dont il faut tenir compte avant d'appliquer ces techniques à de telles flottes d'aéronefs.

Les disques de turbine sont les éléments les plus critiques du point de vue de la sécurité, en raison de l'énergie qui est libérée en cas de rupture. Ainsi, l'amélioration des performances des moteurs à l'avenir est directement liée au degré de résistance aux contraintes de leurs organes. Il faudra de plus répondre à ces sollicitations d'amélioration de performances en respectant les exigences réglementaires en matière de sécurité. Cette situation représente l'un des plus grands défis dans le domaine de la conception des turbopropulseurs.

La satisfaction de ces exigences contradictoires comporte un coût qui dépasse largement les coûts de développement et d'achat d'un moteur particulier. Bien que les disques de turbine soient particulièrement coûteux à fabriquer, plus de 90% de ces éléments sont mis au rebut à moins de 50% de leur vie utile. L'utilisation physique, le cycle de vie des organes moteur et la sécurité doivent être gérés statistiquement en raison des variations des caractéristiques physiques des matériaux.

Étant donnée la prédominance des flottes d'aéronefs en service depuis de nombreuses années, il a été décidé de ne considérer que les aéronefs étant entrés en service avant 1990. Le domaine se trouve ainsi réduit à des solutions pratiques et appropriés pour des exploitants responsables de flottes vieillissantes. Le document demeure toutefois applicable aux nouveaux projets datant de moins de dix ans. L'objectif a été de :

- couvrir les questions techniques et administratives d'une manière qui permet au lecteur de porter son attention uniquement sur les éléments qui l'intéresse. Il s'en est suivi un certain nombre de répétitions entre les différents chapitres;
- Montrer les développements réglementaires parallèles aux Etats-Unis et en Europe;
- Faire la distinction entre les philosophies de conception dite « à vie sûre », et « à tolérance à l'endommagement »;
- Illustrer l'impact des différentes approches en matière de pratique de la maintenance;
- Présenter les résultats obtenus par différents systèmes de contrôle du vieillissement;
- Fournir des conseils pratiques et des directives pour les exploitants de flottes vieillissantes.

Au sein de ce groupe, les principaux concepteurs et gestionnaires de cycle de vie des principaux motoristes dans le monde ont pu travailler avec des représentants de la majorité des exploitants d'aéronefs de l'OTAN.

# Contents

# Members of the AVT-017 Task Group


**Chairman**
Mr. G. Lazalier
Sverdrup Technology, Inc.
872 avenue E
Arnold AFB, TN 37389-5051, USA

**Technical Author and Editor**
Mr. M. Sapsard
Implement Ltd
9 Green Dell Way
Hemel Hempstead
Herts HP3 8PX, UK

**CANADA**

Dr. W. Beres
National Research Council of Canada
1500 Montreal Road, Bldg. M-7
Ottawa K1A 0R6

Mr. D. Rudnitski
National Research Council of Canada
1500 Montreal Road, Bldg. M-7
Ottawa K1A 0R6

**FRANCE**

Mr. C. Colette
Turbomeca
64411 Bordes

Mr. O. Etchevers
SPAe, 26, boulevard Victor
00460 Armees

Mr. P.R. Mosser
SNECMA, Direction Technique
Aerodrome de Villaroche
77550 Moissy Cramayel

**GERMANY**

Dr. J. Broede
MTU München GmbH
Dept. EMB, Postfach 50 06 40
D-80976 München

Prof. K. Boichhausen
MTU München GmbH
Hauptabteilungsleiter
Stromungsmachinenberechnung-EW
Postfach 50 06 40
80976 München

**GREECE**

Capt. D. Adamopoulos
Hellenic Air Force Research Center
Terpsithea Post Office
16501 Glyfada

Prof. P. Kotsiopoulos
Hellenic Air Force Academy
c/o Nazliou 35-39+ Byzadiou 70
17122 - N. Smyrni, Athens

Mr. G. Panagakis
Hellenic Aerospace Industry
P.O. Box 23
Tanagra
32009 Schimatari

**ITALY**

Ing. E. Campo
FIATAVIO s.p.a.
corso Ferrucci, 112
10138 Torino

Mr. C. Vinci
FIATAVIO s.p.a.
via Nizza 312
10127 Torino

**NETHERLANDS**

Mr. L. ten Have
National Aerospace Laboratory NLR
P.O. Box 153
8300 AD Emmeloord

**PORTUGAL**

Lt. Col. M. G. Chambel
CLAFA/DMA
avenida de Forca Aerea
2720 Alfragide

**TURKEY**

Major C. Erel
1nic Hava Ikmal Bakim Merkez
Komutanligi, 26030 Eskisehir

Major E. Gunes
1nic Hava Ikmal Bakim Merkez
Komutanligi, 26030 Eskisehir

**UNITED KINGDOM**

Sqn. Ldr. C. Eady
Logs (OR) le. F10 Royal Air Force
RAF Brampton, Huntingdon
Cambs PE18 8QL

Mr. R.V. Cottington
Propulsion Department
Building 304
DTEO Pyestock
Hants, GU14 0LS

Mr. P. Everitt
Rolls-Royce plc
P.O. Box 3
Filton, Bristol BS34 7QE

Prof. G. Harrison
DERA
Griffith Building (AT)
MSS, Farnborough, Hants GU14 0LX

Group Capt. J.K. Newton
FS(Air) 4, MOD PE (UK)
Maple Oa #45 Abbey Wood, Bristol BS34 8JH

Mr. J. Nurse
FS(Air) MOD PE (UK)
Maple Oa #45 Abbey Wood, Bristol BS34 8JH

Mr. F. Tufnell
FS(Air) MOD PE (UK)
Maple Oa #45 Abbey Wood, Bristol BS34 8JH

**UNITED STATES**
Mr. A. Cifone
Propulsion and Power Engineering
Bldg 106, NAWCAD
22195 Elmer Road, Unit 4.4c
Patuxent River, MD 20670-1534

Mr. O. Davenport
ASC/LP
2145 Monahan Way
Wright Patterson AFB
OH 45433-7017

Mr. R. Holmes
United Technologies - Pratt and Whitney
P.O. Box 109600
West Palm Beach, FL 33410-9600

Mr. P. Maletta
GE Airdraft Engines
Applied Mechanical technology
1 Neuman Way, Mail Stop Q105
Evandale, OH 45215 6301

Mr. C. Meece
Turbine Engine Component Center
Pratt and Whitney Aircraft
P.O. Box 109600, M/S/ 711-31
West Palm Beach, FL 33410-9600

Mr. P. Zimmerman
Naval Air Warfare Center, MS3
Aircraft Div. 22119 James Road
Patuxent River, MD 20670 - 5304

# Contributors

**Chairman:** Glen Lazalier, Sverdrup Technology, AAFB, USA

**Technical author and editor:** Michael Sapsard, Implement Ltd, UK

**Lead authors:**

| | |
|---|---|
| Chapter 1: | Michael Sapsard, Implement Ltd, UK |
| Chapter 2: | Richard Holmes, Pratt & Whitney, USA |
| Chapter 3: | Otha Davenport, ASC/LP, WPAFB, USA |
| Chapter 4: | Christopher Eady, RAF, who succeeded Owen Barnes, RAF, UK |
| Chapter 5: | Wieslaw Beres, NRC, CA |
| Chapter 6: | George Harrison, DERA, UK |
| Chapter 7: | Pierre-Ettienne Mosser, SNECMA, FR |
| | Peter Everitt, Rolls-Royce, UK |
| Chapter 8: | Jürgen Broede, MTU, GE |
| Chapter 9: | Michael Sapsard, Implement, Ltd, UK |
| Appendix 1: | Paul Maletta, GE Aircraft Engines, USA |
| Appendix 2: | Otha Davenport, ASC/LP, WPAFB, USA |
| Appendix 3: | Wieslaw Beres, NRC, CA |

**Contributors:**

**Experts:**

| | |
|---|---|
| Lex ten Have: | National Aerospace Laboratory, NL |
| Paul Zimmerman: | Naval Air Systems Command, USA |
| Carl Meece: | Pratt & Whitney, USA |
| Johnny Adamson: | Pratt & Whitney, USA |
| Dimitris Adamopoulos: | Hellenic Airforce Research Centre, GR |
| Fred Tufnell: | FS(Air), MOD(PE), UK |
| Jim Nurse: | FS(Air), MOD(PE), UK |
| John Newton: | FS(Air), MOD(PE), UK |
| Christophe Colette: | Turbomeca, FR |
| IPA Etchevers: | SPAé, FR |

**Readers:**

| | |
|---|---|
| William Wallace: | NRC, Canada to whom a special debt of gratitude is due. |
| Bernie J Rezy: | Allison USA |
| Kalyan Harpalani | Pratt & Whitney, Canada |

**AVT/Former PEP Panel members:**

| | |
|---|---|
| C Erel: | TU |
| A CiFone: | USA |
| C Vinci: | IT |
| A Kottarakos: | GR |
| R Cottington: | UK |
| D Rudnitski: | CA |

# Publications of the RTO Applied Vehicle Technology Panel

MEETING PROCEEDINGS (MP)

**Design for Low Cost Operation and Support**
MP-37, Spring 2000

**Structural Aspects of Flexible Aircraft Control**
MP-36, Spring 2000

**Aerodynamic Design and Optimization of Flight Vehicles in a Concurrent Multi-Disciplinary Environment**
MP-35, Spring 2000

**Gas Turbine Operation and Technology for Land, Sea and Air Propulsion and Power Systems (Unclassified)**
MP-34, Spring 2000

**New Metallic Materials for the Structure of Aging Aircraft**
MP-25, April 2000

**Small Rocket Motors and Gas Generators for Land, Sea and Air Launched Weapons Systems**
MP-23, April 2000

**Application of Damage Tolerance Principles for Improved Airworthiness of Rotorcraft**
MP-24, January 2000

**Gas Turbine Engine Combustion, Emissions and Alternative Fuels**
MP-14, June 1999

**Fatigue in the Presence of Corrosion**
MP-18, March 1999

**Qualification of Life Extension Schemes for Engine Components**
MP-17, March 1999

**Fluid Dynamics Problems of Vehicles Operation Near or in the Air-Sea Interface**
MP-15, February 1999

**Design Principles and Methods for Aircraft Gas Turbine Engines**
MP-8, February 1999

**Airframe Inspection Reliability under Field/Depot Conditions**
MP-10, November 1998

**Intelligent Processing of High Performance Materials**
MP-9, November 1998

**Exploitation of Structural Loads/Health Data for Reduced Cycle Costs**
MP-7, November 1998

**Missile Aerodynamics**
MP-5, November 1998

EDUCATIONAL NOTES

**Measurement Techniques for High Enthalpy and Plasma Flows**
EN-8, April 2000

**Development and Operation of UAVs for Military and Civil Applications**
EN-9, April 2000

**Planar Optical Measurements Methods for Gas Turbine Engine Life**
EN-6, September 1999

**High Order Methods for Computational Physics (published jointly with Springer-Verlag, Germany)**
EN-5, March 1999

**Fluid Dynamics Research on Supersonic Aircraft**
EN-4, November 1998

**Integrated Multidisciplinary Design of High Pressure Multistage Compressor Systems**
EN-1, September 1998

**TECHNICAL REPORTS**

**Recommended Practices for Monitoring Gas Turbine Engine Life Consumption**
TR-28, April 2000

**Verification and Validation Data for Computational Unsteady Aerodynamics**
TR-26, Spring 2000

**A Feasibility Study of Collaborative Multi-facility Windtunnel Testing for CFD Validation**
TR-27, December 1999

# Chapter 1

## Introduction to Engine Usage Monitoring

by

*(M. Sapsard)*

**Page**

# 1. INTRODUCTION

Engine usage monitoring systems exist for the following reasons:

- To allow aircraft to be operated safely;
- To ensure that as much life as possible is extracted from life limited parts, hence reducing costs;
- To allow performance levels to be pushed to higher levels with lighter, smaller components;
- To provide information for next generation engine design.

This is because, if the right counter-measures and safety precautions are not in place, the highly stressed components in modern engines can fail in service - catastrophically, and without warning. These preventive processes begin during design and continue throughout the life of an engine. The failure mechanism of most concern, and discussed in detail, is low cycle fatigue.

We aim to show what is being done and describe in detail the fundamentals of engine monitoring systems. After reading this document, managers of older aircraft and specifiers of new aircraft should be in a good position to ask the right questions and make the right decisions about the systems they have, should have, or will have.

# 2. LIFING PHILOSOPHIES

There are two main approaches to the design of life critical parts in gas turbines. In all components, metallurgical defects may become the initiation point for a crack. These may in turn lead to component failure. The first approach aims to retire components before the crack is detectable. The second approach tolerates a growing crack and aims to remove the component when the crack is detectable. In theory, crack growth may be monitored. In general practice, safety-critical components are removed as soon as a crack is detected.

## 2.1. 'SAFE LIFE' DESIGN

The first lifing approach is to assume that when a crack appears, the component has failed, and to ensure that all components are retired before the first crack appears. This clearly carries a penalty in that many components will be retired prematurely. This approach is known as 'Safe Life' or 'Life to First Crack'. The lifing method employed for these parts follows a "cycles-to-crack initiation" criterion where a minimum life capability is defined statistically for simulated service conditions through extensive coupon testing and component test verification in a spin pit test rig. The statistical minimum is usually based on the probability that no more than, say, 1 in 1000 components will have developed a detectable crack (typically chosen as a crack of 0.8 mm in length or 0.38mm in depth). It should be mentioned that the value '1 in 1000' is chosen here arbitrarily. Different manufacturers use different values that depend on the risk that they are prepared to take. For example, 1 in 980, or 1 in 750 components.

## 2.2. 'DAMAGE TOLERANCE' DESIGN

The second lifing approach is to design components so that they are tolerant of cracks. However, components must still be retired before they become dangerous. This approach carries a weight penalty and the additional costs of a highly structured and organised inspection system. This is known as the 'damage tolerance' or '2/3 dysfunction' approach. This lifing method demonstrates that the component is capable of continued safe operation during crack growth. This is if the cracks grow sufficiently slowly during service to allow their growth to be reliably detected, and perhaps even monitored, through regularly scheduled inspections.

Both methods aim to provide similar safety factors, and frequently use very similar detail design techniques. The cost effectiveness of each approach depends on the industrial and operating environment in which the equipment is designed, built, operated and maintained.

Associated terms, which may cause confusion, are:

- Fracture mechanics - the science of predicting crack growth;
- ENSIP - the engine structural integrity program, which is a US attempt to close the design loop through extensive component testing to the point of failure.

ENSIP embraces the damage tolerance approach. Only by testing until components fail is it possible to be certain of their characteristics and failure modes. If the tests were only conducted on laboratory models, the results would be more dependent on the assumptions made. Because components are regularly inspected for cracks, and the crack growth may be monitored, the loop is closed throughout the life of the fleet.

Damage tolerant design does not necessarily mean that an ENSIP type of program must be followed. In the UK, a database approach to materials characterisation has been used successfully for components that have been designed to be damage tolerant.

# 3. SCOPE

Whichever lifing policy is used it is necessary to know how engines are used in service, in order to calculate how much life has been consumed. This can be done either by monitoring the usage of a representative sample of the fleet and hence calculating the fleetwide usage or by monitoring engines on an individual basis and treating them accordingly. The latter approach requires a higher initial investment, but allows a higher proportion of life to be consumed safely. This is largely because there is no need to allow for the statistical variation between the sample and the fleet. This aspect is described in detail in chapter 9.

Monitoring systems depend on the design data and rules of the systems they monitor. This knowledge was used by the working group to categorise engines into four generations. All generations are in use and will continue to be in use for many years. It should be noted that many real life systems inevitably straddle the generations that are loosely defined below, but this did not reduce the usefulness of the categories as working group guidelines.

This categorisation had a major influence on the scope and arrangement of the subsequent work. It is included here because it provides historical context for the engine lifing methods that are in use today, and a justification for the self imposed constraints that the group agreed.

The final scope of the work included lifing and usage monitoring of gas turbine engines that were designed up to the middle or end of the 1980s. How this was decided is indicated below.

## 3.1. CLASSIFICATION OF ENGINES

### 3.1.1. FIRST GENERATION

- Simple materials data is used;
- Simple design rules are used - disks and blades only;
- Components are lifed to first crack using primitive crack detection techniques (hot oil and chalk);
- Simple mission profiles are used for design purposes;
- Broad usage assumptions are made in terms of so many cycles per hour;
- Life is measured in operating or flying hours;
- Fleetwide life is extended by inspection of lead engines.

### 3.1.2. SECOND GENERATION

- Design data and rules remain unchanged;
- Spin pit testing is introduced;
- Tape based recorder systems are used;
- Most of the available aircraft and engine data is recorded;
- Only a small part of the fleet is monitored;
- Ground based analysis is done at the manufacturer's plant;
- Only small sample data is available to help assess hourly usage rates.

### 3.1.3. THIRD GENERATION

- Monitoring equipment is fitted to all aircraft/engines in the fleet;
- Lifing by life units is converted to hours to conform to the airworthiness documents;
- Data is analysed on-board;
- Flight by flight results are downloaded;
- Fleet usage management is by component rather than engine;
- Tracking of individual parts is normal;
- Data is used to feed into new engine specifications;
- Some fracture mechanics methods may be used.

### 3.1.4. FOURTH GENERATION

- Design is fully dependent on the materials database;
- Fracture mechanics based design techniques are extensively used;
- Comprehensive time domain stress models are developed;
- Life is to 2/3 dysfunction or limiting acceptable crack length or depth;
- Every critical feature is individually tracked;
- Lifing is by life units that are not hours;
- The monitoring system is integrated into the engine design and development process.

## 3.2. SCOPE OF EFFORT

To see whether there were any underlying systematic characteristics, the engines known to those present were considered in two ways. Firstly by 'start of design' date, and secondly by the technology used to build the engine. The results of both are shown below.

### 3.2.1. ENGINES IN USE

An examination of engines introduced into service since the 1950s produced figure 1. The dates shown are the estimated 'start of design' dates which was taken to be a better indicator of design philosophy than 'entry into service'. Because of the commercial competition and proprietary information surrounding the 1990s engines, these were excluded from further consideration. All of the engines are believed to be still in service. This suggests that some engines are likely to achieve much longer service lives than were ever imagined when they were conceived. Indeed some have had the life of their critical components re-estimated using modern design tools.

### 3.2.2. TECHNOLOGY

Consideration of the control technology used and the differing maintenance and economic drivers led to the generation of figure 2. The four generations were typically identified and decisions made as to which generations should fall within the scope of the report.

| 1950s | 1960s | 1970s | 1980s | 1990s |
|---|---|---|---|---|
| J57 | TF41 | F100 | F110 | EJ200 |
| J75 | J85 | F101 | F100 | F414 |
| TF33 | TF30 | F108 | F200 | M88 |
| Dart | TF34 | CFM56 | F220 | |
| Conway | TF39 | T700 | F118 | |
| T56 | Spey | Makila | AE2100 | |
| J69 | Adour | TFE125 | RTM322 | |
| Tyne | Pegasus | RB199 | F117 | |
| J79 | Gem | F404 | F119 | |
| T58 | Turmo | M53 | F120 | |
| | Astazou | RD33 | MTR390 | |
| | ALF502 | PT6 | | |
| | Allison 250 | | | |
| | T64 | | | |
| | ATAR | | | |
| | Larzac | | | |
| | Gnome | | | |

Figure 1 – Engine generations by design year

Engines that started design between approximately 1950 and 1989 were considered 'In-scope'. This encompassed all of the old, mid-life and new engines operated outside of their home territories, but covers most practices up to the current time.

## 4. MAXIMISING THE BENEFITS OF USAGE MONITORING

To know the life remaining for each component is clearly the first step towards being able to run each component in service for as long as possible. A manual log might be sufficient for the operator to be in full control. However, in a busy operational environment the management problems are complex. Aircraft come and go and engines are frequently managed on a modular basis.

| | 1950s | 1960s | 1970s | 1980s | 1990s |
|---|---|---|---|---|---|
| **Control** | Mech | Hydro/ Mech | Electro/ Mech | Elec/ Mech/ Fadec | Fadec |
| **Monitor** | Manual | Manual EHM | EHM | EHM | EHM |
| **Maint. Policy** | Hard Life | Hard Life | Hard Life/ RFC | FM RFC Hard Life | FM RFC |
| **Drivers** | Safety | Safety | Safety/ Economic | Safety/ Economic | Safety/ Economic |
| **Generation** | 1 | 1 | 2 | 3 | 4 |
| **Report** | Old Engines | | New & Mid-Life | | Future |
| **In Scope** | Yes | | Yes | | No |

RFC - Retirement for cause.
FM. - Fracture Mechanics.
FADEC - Full Authority Digital Engine Control.
DT - Damage Tolerance

Figure 2 – Concurrent technology and maintenance philosophies

This means that a component might be flying in three different aircraft at three different bases within a period that is as short as six months. The only way to manage this is to run a comprehensive parts life-tracking system.

To put the frequency of engine changes in a mature fleet into perspective, figure 3 shows Adour Mk 104 engine rejections from the RAF Hawk fleet over a sixteen-month period. 43% were due to cracking or engine component life limits. This indicates that engine removals to replace life expired components is a major nuisance and cost driver in mature engine fleets. Every fleet manager would presumably welcome the tools for managing component lives as effectively as possible.

Just by using hourly lifing methods, it is possible to create charts like figure 4. That shows the remaining life for various turbine blade sets. As the remaining life reduces, the distribution for each group spreads. This illustrates a phenomenon that is typical of many mechanical systems.

By adding more information, namely the usage rate for each engine, figure 5 can be constructed. The example is for Adour Mk 104 HP Turbine disks. The significance of this chart is that a tidal wave of life expired components is only 4 to 5 quarters away. With a lead-time of two to three years for new components, this type of information is essential for the supply organisations.

This approach is of value to both first line and logistic planning, At first line aircraft can be assigned duties so that availability is maximised. At HQ, purchase orders can be placed, so that sudden increases in the number of life expired components do not ground the fleet.

To maximise these gains it is important that the lives of the many components on an engine are synchronised to maximise the on-wing period. Tools for doing this already exist, and examples are the UK Pegasus (PRAMS) system, in figure 5, and the GE Tornado OLMOS system, which is shown below, in figure 6. Once tools like these are in place, the full benefits from individual component monitoring on a fleetwide basis can be achieved.

Figure 6 shows information about engine number 4029, its position, its running time and its flight time. Life remaining for each of the main stress features in the low-pressure compressor (NDV), intermediate compressor (MDV), the high-pressure compressor (HDV), and the

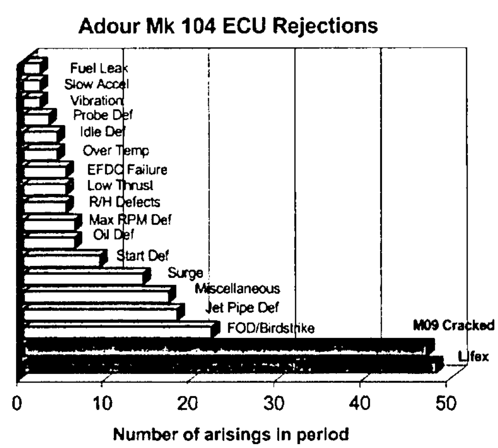



Figure 3 – Typical engine rejections

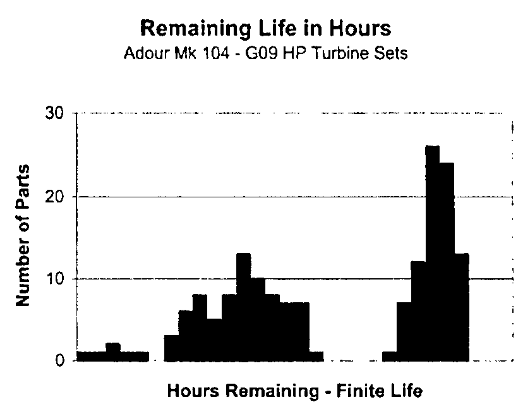



Figure 4 – Typical component life distribution

corresponding turbines are also shown. A graphical display of the remaining life, compared to the fleet average is also shown.

## 5. COST SAVINGS

Although every costing exercise is unique due to differences in accounting, operational and commercial practices, some cost studies have been done on retrospective fitting of life monitoring equipment. Figure 7 shows the estimated benefits that will be enjoyed by the RAF as a result of fitting simple engine life recorders to the RB199 engines in the ADV Tornado aircraft. The algorithms are speed related only, with engine speeds sampled eight times per second.

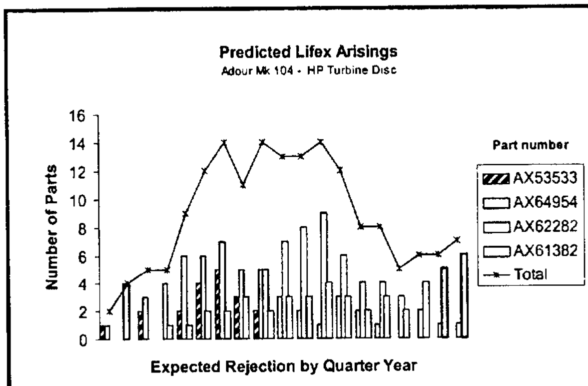



Figure 5 – Typical component life expiry

Figure 7 clearly puts the time-scales for adding monitoring systems to ageing fleets into perspective. In this case, the financial benefits were about £20M after thirteen years, and the breakeven point was about eight years from the beginning of the project. The slowdown in 2006 is due to replacement of some aircraft by Eurofighter. This takes no account of the safety benefits that should accrue from a better knowledge of fleet operation. Even if this amount of money appears trivial, to dismiss other programs as not worthwhile, because of this cost study, would be shortsighted. Every situation should be treated on an individual basis.

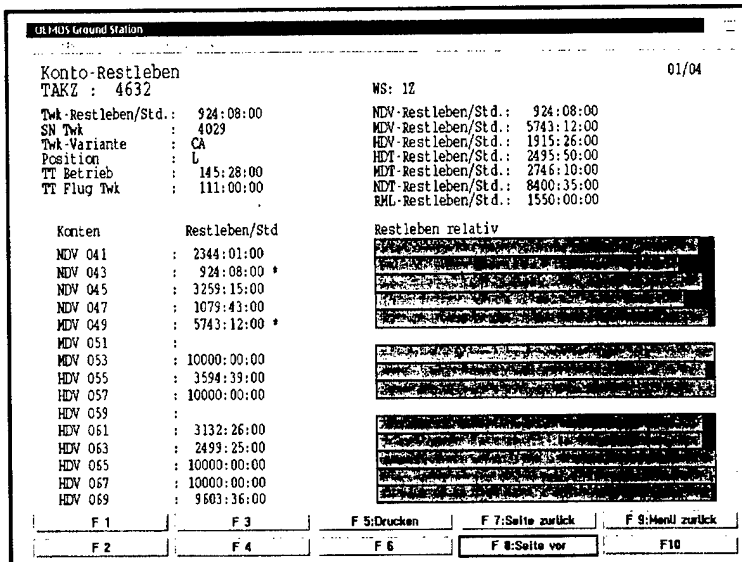



Figure 6 – Typical display of engine usage information

The thirty-year old aeroplane is a fact. When purchases are made, of new or used aircraft, monitoring systems should be included in the negotiations. If sampling is considered, then figure 7 indicates that a full fleet fit saves more than twice as much as a half-fleet fit over just thirteen years. Referring to figure 1, it may be seen that some engines may remain in service for 40 to 50 years! Therefore, the benefits that should accrue from a modern monitoring and management system, especially if fitted before entry into service, should be considerable. Strong supporting evidence for this is in chapter 3.

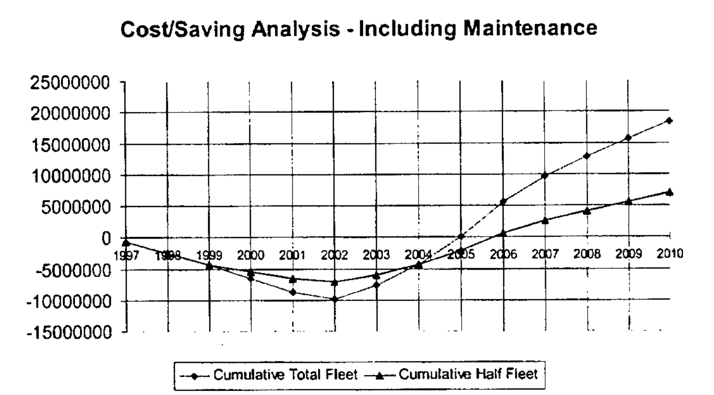



Figure 7 – Cost benefit analysis for simple life counters on Tornado

## 6. SUMMARY

This introduction has attempted to provide an overview of the safety and financial requirements that make the subject of component life monitoring so important. The topics touched upon have been diverse, with the intention of providing an appreciation of the breadth of the subject.

In the following chapters, many different aspects of life usage monitoring are discussed in detail, ranging through regulatory, managerial, technical and operational perspectives. Here only the surface has been touched.

## 7. REFERENCES


*The Story – People and Eucams.* Jan 1994, ISBN 09522854 01. Published by Future Systems (Air)42, MOD(UK).

Figure 6 – Courtesy of MTU.

Figure 7 – Courtesy of Rolls-Royce and FS(Air)42, MOD(UK)

# Chapter 2
## Civil and Military Practices

by

*(R. Holmes)*

# 1. INTRODUCTION

Design practices, life prediction methods, life usage monitoring systems, and control systems for gas-turbine engines have all improved since they were in their infancy in the 1940's. Then, a 25-hour test was the main pass or fail criteria for a new engine entering service. As with many engineering practices, the developments evolved in response to durability problems and a desire to do a better job of producing and maintaining gas turbine engines.

The need for engine reliability and durability was recognised in the US MIL-E-5007C specification, as long ago as 1966. Although lacking the rigorous analysis, test and demonstration aspects of today's specifications, MIL-E-5007C called for a reliability of 500 hours between power loss events, and an inherent engine life of 5000 hours within the specified environment. MIL-E-5007C also required a 150-hour life demonstration when tested in accordance with the MIL-E-5009 specification. In the UK, D. Eng RD 2100 and 2300 provided design requirements and guidance for all aspects of gas turbine development, from the early 1950's until very recently. As time passed these increasingly took account of the practices adopted in the US.

Today the 'Safe-life' and the 'Damage Tolerance' approaches are the two most widely used design methods for producing components that meet life requirements. The processes which are codified in specifications such as MIL-STD-1783 (ENSIP), MIL-E-5007E, FAR 33, MIL-E-8593A and JAR-E cover transport, helicopter, and fighter engines. Their scope addresses design, but also material characterisation, engine testing, and gathering of information on usage, maintenance and inspection. In this chapter the evolution of engine controls, the regulations governing structural integrity and methodologies for determining the life of components that ensure gas turbine engine durability are discussed.

## 1.1. CONTROL SYSTEMS TECHNOLOGY

In chapter 1, the technology factors that were used to decide the scope of this document were discussed. One of the major enabling factors for real-time present day systems is the ready availability of the necessary data from the instrumentation that already exists for engine and aircraft control purposes. Hence, it is relevant to consider control systems and their development to the current state of the art.

### 1.1.1. HYDRO-MECHANICAL DEVELOPMENT

The 1950's and 1960's saw an era dominated by the use of hydro-mechanical controls which demonstrated reliability rates as high as 300,000 hours between in-flight-shutdowns. Engines did not need the precise or high-speed response of electronic controls, which was rather fortunate!

The basic function of modern electronic engine control systems is to adjust variable geometry, control fuel flow, trim, and regulate other actions such as the application of anti-icing systems. The logic for controlling these functions varies with throttle movement, position within the flight envelope (Mach number, temperature, pressure, altitude, etc.), and engine requirements to provide service bleed air. This data can be used to provide automatic thrust rating control, engine limit protection, transient control and engine starting. These electronic systems are simpler to operate by the flight crew, save fuel, are easier and less expensive to maintain, and improve reliability standards when compared to the hydro-mechanical controls of the past. The implementation history of electronic controls at, for instance, Pratt and Whitney dates back to the 1950's when an early attempt was made, using vacuum tube technology, to develop a system for the J57 engine. The harsh vibration and acoustic environment proved too much for these early controls.

### 1.1.2. ANALOGUE ELECTRONIC CONTROLS

The Rolls-Royce Gnome helicopter engine (developed under licence from the General Electric T58) entered service in 1961 with an analogue electronic control system that was designed to reduce the helicopter pilot workload. The system primarily governed the power turbine output shaft speed by controlling engine fuel flow. It also provided temperature limiting, droop slope matching, and throttle rate control functions. If the electronic system failed, the control functions reverted to the pilot, who controlled engine fuel flow using information from the cockpit gauges.

The Concorde aircraft with Rolls-Royce and SNECMA Olympus 593 engines has a two-lane full-authority analogue engine control as the original design standard. The first flight was in 1969 with entry into service in 1976.

### 1.1.3. DIGITAL ELECTRONIC CONTROLS

The advent of monolithic integrated circuits that were developed in the mid-1960's addressed control system reliability problems and made electronic controls more practical. In 1968 Pratt and Whitney was competing for the FX engine program with General Electric and elected to use electronic technology in the proposed engine. Because the reliability of electronic technology had not yet been proven, the engine was to operate with hydro-mechanical controls and would include a supplementary electronic unit.

The hydro-mechanical unit controlled all engine functions while the electronic unit, known as the supervisory control, operated several electric motors that trimmed exhaust gas temperature and engine speed to maximise performance. Supervisory control systems have an electronic unit, which is superimposed on a hydro-mechanical system, to govern power management. If the electronic unit fails, the system reverts to hydro-mechanical control.

In the 1970's, the United States government funded electronic control research, culminating in the Digital Electronic Engine Control (DEEC) which was incorporated on the F100-PW-220 in 1985 (Kuhlberg et al, 1989). This full-authority electronic control for military engines was two and half times more complex than the preceding supervisory control, and cost the same as the control system that it replaced.

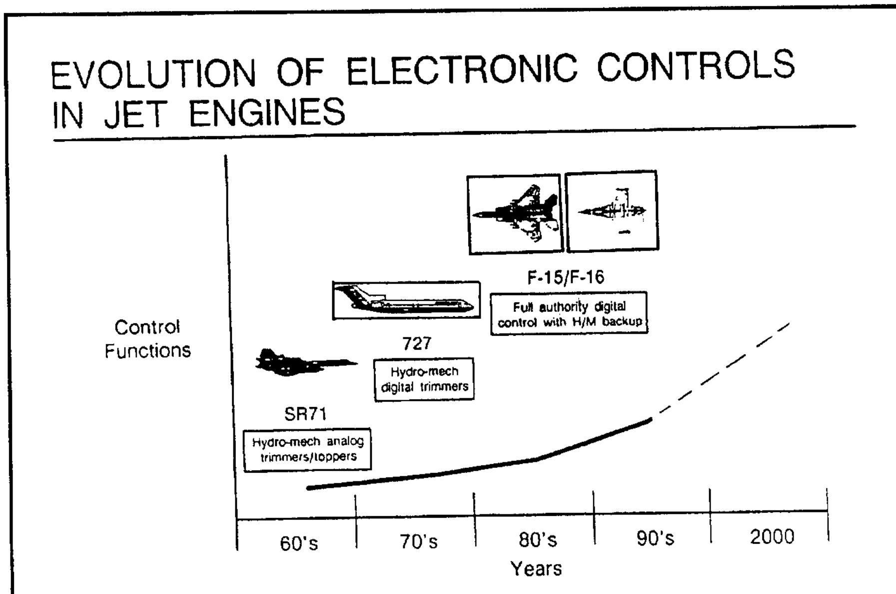



Figure 1 - Evolution of Electronic Controls

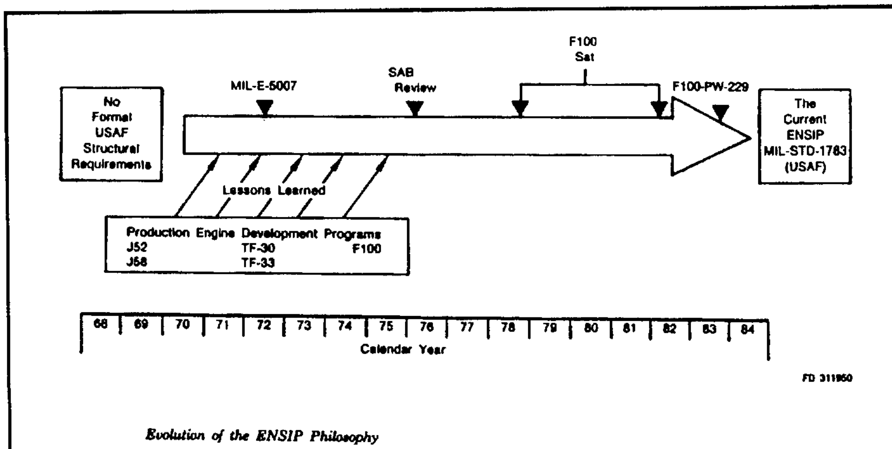



Figure 2 - Changing US Durability Requirements

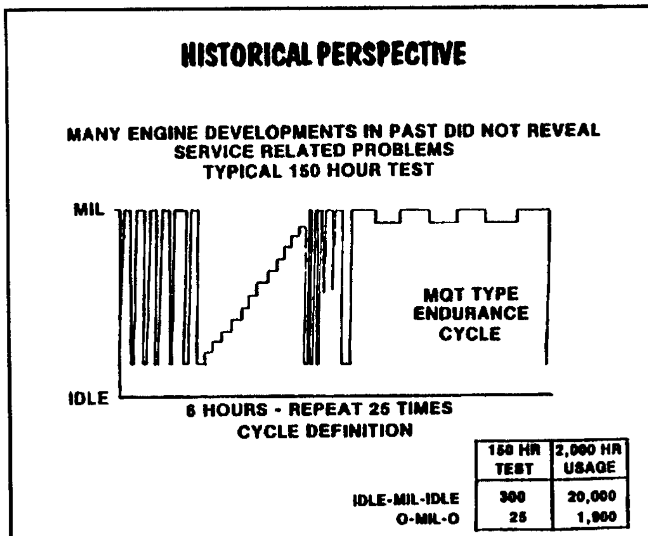



Figure 3 Typical 150-Hr test

Full authority control systems perform all control functions electronically and do not interact with a hydro-mechanical system. Each of these electronic systems had a single set of sensors and computers. If a disabling failure occurred in any of these elements, the system reverted to a hydro-mechanical operation. Even the DEEC can revert to a simple backup mechanical system in the advent of a failure.

Digital Electronic Engine Control systems were flight tested on Rolls-Royce Gem helicopter engines from 1979 and entered service in 1985. These systems normally provided engine control and governing of the power turbine (and aircraft rotor) speed. If a system failure occurred, the pilot took manual control of engine fuel flow.

The military efforts spawned commercial efforts, which in the US featured a capability demonstration and a reliability demonstration. The capability demonstration was carried out in 1975 on a JT8D engine and in 1978 on a JT9D engine. Both the JT8D and JT9D systems were supervisory electronic plus a hydro-mechanical control. The reliability demonstration was carried out on several, airline operated, Boeing 727 aircraft equipped with JT8D engines. While the electronic system did not control any engine functions, it did carry out error detection and diagnostic tests in a closed loop system by addressing outputs and inputs. Over a six-year period covering 400,000 flying hours, these electronic controls demonstrated reliability levels of 20,000 hours mean time between failures (Kuhlberg et al, 1989).

A full authority, dual channel, Engine Electronic Control (EEC) was certified on a PW2037 engine in 1983. Electronic engine control is a major advance in gas-turbine engine control-systems that provide significant benefits in operability, supportability, reliability and safety over hydro-mechanical controls. Figure 1 shows how the growth of control functions has increased over time.

### 1.1.4. Coincident Development of ENSIP and Electronic Controls

The first implementation of full digital electronic engine controls roughly coincided with the introduction of United States Air Force MIL-STD-1783 or (ENSIP). ENSIP was issued in 1984 and digital

controls were introduced on the F100-PW-220 in 1985. Both efforts aimed at improving engine reliability.

# 2. Structural Integrity Requirements

## 2.1. Airframes - US

The United States Air Force approach to structural-fatigue certification was outlined during the 1959 Symposium on Fatigue of Aircraft Structures. Because of the increasing number of fatigue failures/incidents that occurred during the 1950's, a large-scale effort was conducted in 1958 to determine the steps that were necessary to improve aircraft service life. This culminated in the Aircraft Structural Integrity Program (ASIP) which was published in 1969 and contained changes in the following areas:

- Structural development process;
- Structural testing practices and policies;
- Monitoring the actual aircraft usage;
- Air Force Fatigue Certification Program.

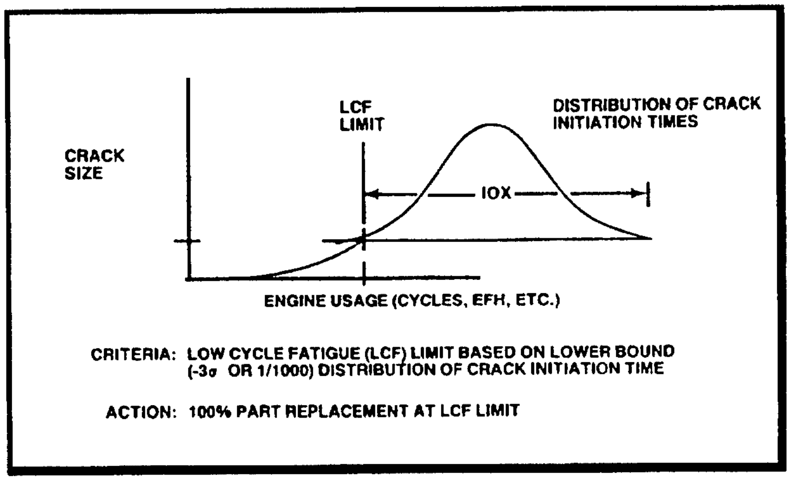



Figure 4 – Safe life approach

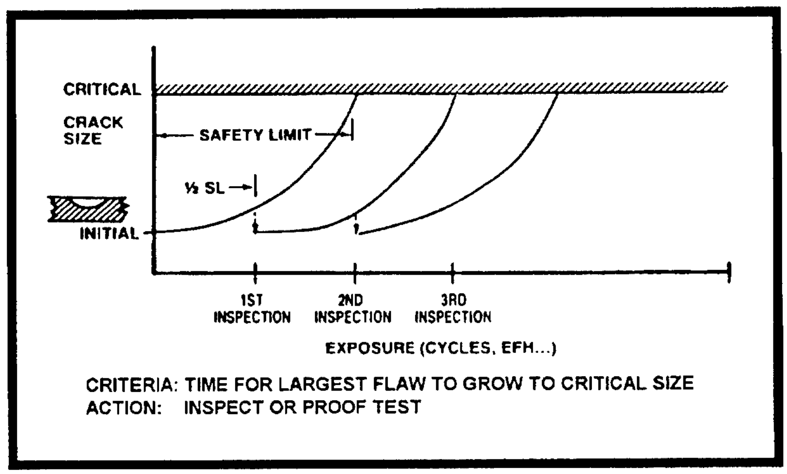



Figure 5 – Damage tolerance approach

## 2.2. Engines - US

Engine requirements for structural integrity followed a similar path to those of the airframe program, a short while later. Figure 2 shows how the engine durability requirements came about for US Air Force gas-turbine engines. Prior to 1969, early structural assessments were based on engine testing which in 1946 was 25 hours and increased to 150 hours in 1952. This cycle was very simple, as shown in figure 3, and unfortunately, it did not reveal service-related problems. Before 1969 engine specifications were deficient in such areas as life requirements, duty cycles, analysis, and testing that was not mission based.

Lessons were learned by reviewing the TF30 and other engine designs and in 1973, the U.S. Navy issued the MIL-E-5007D specification, which established rigorous standards for ensuring engine durability. This document included the *Safe Life* concept, and recognised that components should be retired at the minimum calculated and validated fatigue life. See figure 4.

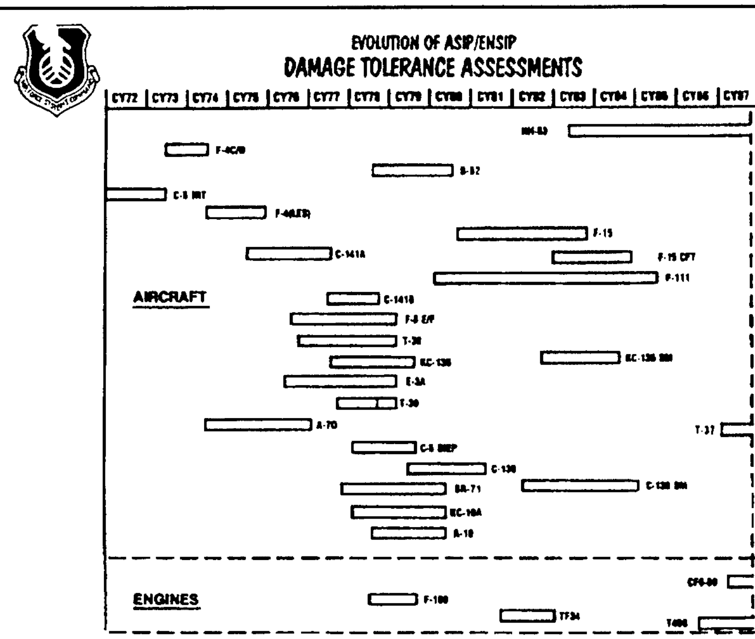



Figure 6 – Evolution of ENSIP

Commercial regulations for gas-turbine reliability practise the safe-life approach. Review of these regulations is pertinent to the discussion of military transport, helicopter, and fighter applications as many of these commercial engines have found their way into military applications. In some cases, as discussed elsewhere, engines that were designed to civil specifications are further scrutinised under a separate and quite different set of military standards.

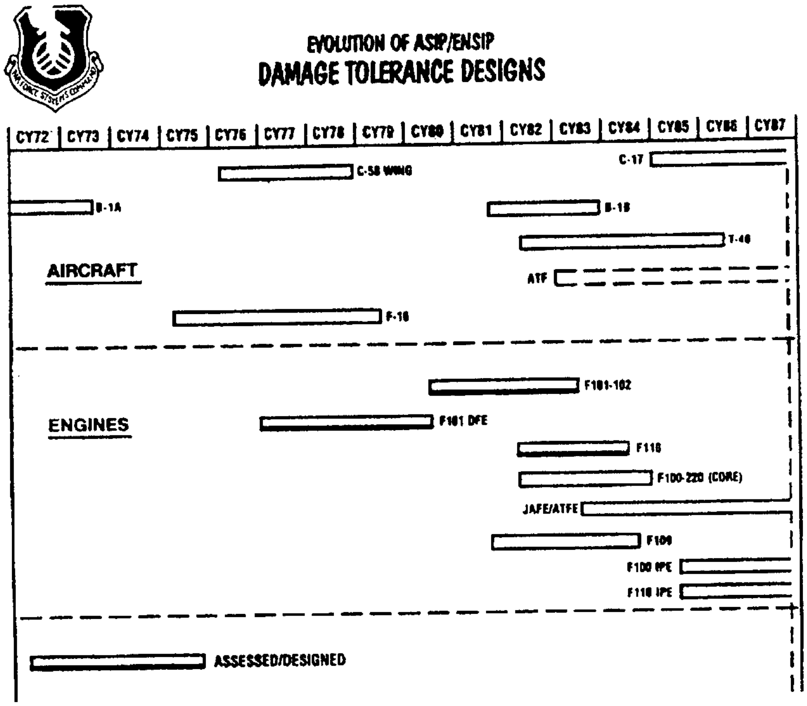



Figure 7 – Application of damage tolerant design

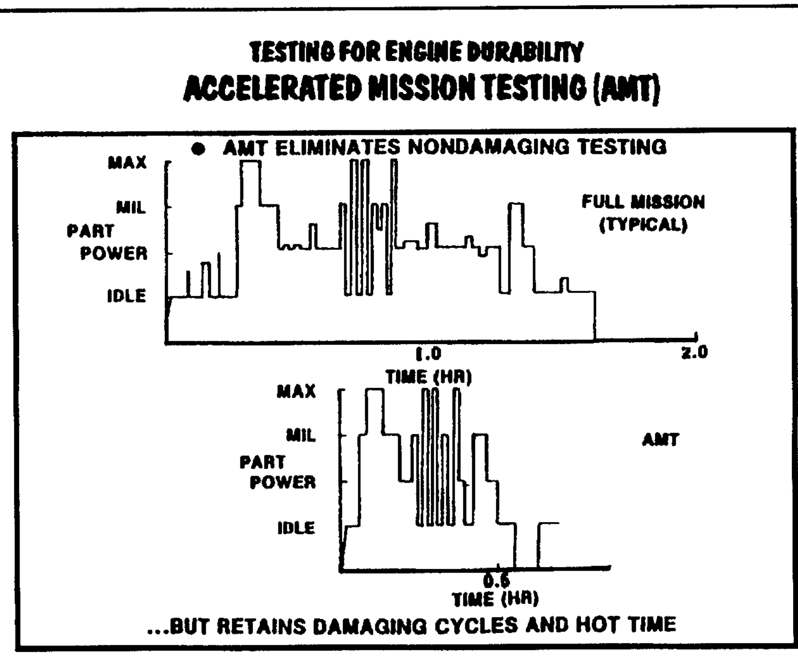



Figure 8 – ENSIP Accelerated Mission Testing

## 3. Regulatory Developments

The history of regulatory developments is:

- Prior to 1965, the civil regulations, both the US Civil Air Regulations (CAR) and the British Civil Air Regulations (BCAR), dealt with durability issues through testing.
- In 1965, in the United States, the Federal Air Regulations (FAR) were codified. These defined operating limitations for disks and spacers in terms of service life.
- By 1970, FAR 33.14 placed restrictions on allowable life of a component through test demonstration.
- In 1976, a Scientific Advisory Board was convened to review engine durability issues. Findings from the panel included:

1. '...we need to apply a system of discipline to our development process';
2. '...Air Force should define an aggressive program for engine mechanical and structural integrity and durability';
3. '...this program should be required by regulation';
4. '...durability and damage tolerance assessment should be performed on fleet engines analogous to those being performed on several weapon system airframes;'
5. By 1980, the FARs permitted approved procedures to be used. These covered both analytical methods and testing. In Europe, the BCARs were combined into the Joint Air Regulations-E (JAR-E) during a similar period and Change 9 permitted analytical life prediction.

### 3.1. Damage Tolerance

*Damage Tolerance* is the ability of a component to resist failure due to the presence of flaws, cracks, or other damage. This method is illustrated in figure 5. Following the recommendations of the advisory board data was collected from various engines and structural assessments were conducted to determine what the impact of applying the damage tolerance procedures would be. These structural assessments included the following:

- Damage tolerance assessments of components previously not reviewed with fracture mechanics analysis;
- Updated low cycle fatigue analysis with improved mission definitions;
- Evaluation of inspection procedures and their implementation;
- Establishment of maintenance plans that incorporate non-destructive evaluation;
- Development of improved durability manufacturing methods.

Other related efforts concentrated on the generation of crack growth data on titanium and nickel base alloys, and the testing of cracked components in spin rigs and in engines. In 1984 the Engine Structural Integrity Program Standard (MIL-STD-1783), which is the current standard for designing and maintaining gas turbine engines utilising damage tolerance procedures, was released by the United States Air Force. Some airframes and engines assessed with the damage tolerance process and those that have been subsequently designed with these procedures are illustrated in figures 6 and 7 (Ogg, 1998).

## 3.2. DEVELOPMENT OF GENERAL PROCEDURES

A brief overview of each of the military and commercial practices used to produce durable gas turbine designs for military fighter, helicopter, and transport applications follows. Many have common features.

### 3.2.1. MIL-E-5007E (MILITARY SPECIFICATION)

The current 5007E specification was issued in 1983 and superseded 5007D, which was issued in 1973. This was the original military specification that placed strict cyclic life requirements, maintenance, testing, and design processes on gas turbine rotating hardware. It is most widely used by the US Navy. The 5007E specification requires that the safe-life approach be practised on critical parts. These parts are minimally defined as the engine pressure casing and all rotor stages.

Military specification MIL-E-5007E includes the following design requirements.

- All engine critical parts are to be designed to twice the life requirement.
- Conventional structural criteria are defined to ensure durability from such failure modes as burst strength, vibration, low cycle fatigue, high cycle fatigue, and creep.
- Materials are to be characterised with a three-sigma standard deviation, and fracture toughness of the material is to be considered.
- Consideration is to be given to engine-airframe inter-actions.

Military specification MIL-E-5007E Structural requires verification testing and measurements as follows:

- Turbine and compressor rotor overspeed;
- Over-temperature of the first stage turbine rotor by 81 degrees Fahrenheit;
- Disc burst testing of all rotating discs to a minimum of 122 percent of the maximum allowable steady-state speed;
- Engine static load tests;
- Substantiation of durability with a Durability Proof Test consisting of 300 hours of accelerated mission-oriented endurance testing that is preceded and followed by stair-step/Bodie test schedules.
- Substantiation of durability by Accelerated Simulated Mission Endurance Testing (ASMET) equal to at least 1000 hours or one half of the cold parts lives;
- Low cycle fatigue testing of three sets of critical engine components via spin pit testing.

Components are monitored in the field, design mission profiles are validated and the operating conditions are recorded. By monitoring this operating experience, and providing feedback into the life analysis process, new lives are determined for components in a service environment. To further improve life assessments, inspections of fracture-critical in-service parts are carried out on an opportunistic basis.

### 3.2.2. MIL-STD-1783 (ENSIP)

MIL-STD-1783, the Engine Structural Integrity Program (ENSIP), was issued in 1984 by the United States Air Force and followed the development of MIL-E-5007 to which it has many similarities. ENSIP is an organised and disciplined approach to the structural design, analysis, development, production, and life management. Areas stressed in ENSIP include:

- Development plans;
- Operational requirements;
- Design analysis, material characterisation, and development tests;
- Component tests;
- Core and engine tests;
- Ground engine tests;
- Flight engine tests;
- Engine life management.

ENSIP emphasises the damage tolerance approach, with crack-growth resistance-criteria added to maintain reliability in the presence of manufacturing and processing defects in the materials. Components are classified into the following categories:

- *Safety Critical* for components that will result in the probable loss of the aircraft, or a hazard to personnel due to direct part failure or by causing other progressive part failures;
- *Mission Critical* for components whose failure will generate a significant operational impact by degrading mission cap-ability to the extent of creating an indirect safety impact on the aircraft. Items whose failure will cause a non-recoverable in-flight shutdown on multi-engined aircraft are considered mission critical.
- *Durability Critical* components are functionally or structurally significant, and in the event of failure possess the potential for generating a significant economic impact but will not necessarily impair

safety or mission capability;

- *Durability Non-critical* component failures result in a minor economic impact or minor degradation of mission capability, and may initiate unscheduled maintenance actions;
- *Other.*

Other key features of MIL-STD-1783 include the following, many of which already existed in MIL-E-5007E:

- Structural design must be based on realistic mission usage and service life requirements;
- Conventional structural criteria are issued to ensure durability from such failure modes as vibration, low cycle fatigue, high cycle fatigue, and creep;
- Damage tolerance (crack growth resistance) criteria are added to maintain reliability in the presence of materials, manufacturing, and processing defects. Parts classified as safety or mission critical must receive damage tolerance analysis.
- Compliance may be achieved through analysis or verification;
- New materials and processes are characterised;
- The internal operating environment must be defined from analysis and engine thermal/performance test data;
- Durability must be substantiated by engine testing or an Accelerated Mission Test (AMT) as illustrated in figure 8. As the mission and engine testing progresses the design is released for increasing levels of use, as illustrated in figure 9.

### 3.2.3. MIL-E-8593A (MILITARY STANDARD)

The general specification for aircraft with turbo-shaft and turbo-prop engines was released in 1975. Its preamble states 'This specification establishes the performance, operating characteristics, design features, detailed interface configuration definitions, and installation envelopes for turbo-shaft and turbo-prop engines. It also establishes the demonstrations, tests, reports, inspection procedures, and other data required for satisfactory completion and acceptance, by the using service, of the Preliminary Flight Rating Tests (PFRT) and the Qualification Tests (QT) for the engines.' MIL-E-8593A mirrors the structural integrity requirements of Mil-E 5007E. The differences between the two documents exist in paragraphs pertaining to requirements other than life or structural performance.

### 3.2.4. FAR 29 - CIVIL STANDARD

FAR 29 or 'Airworthiness Standards -Transport Category Rotorcraft Sub-part E' refers to powerplants. Section 29.903 discusses the engines and the need to minimise hazards to the rotorcraft in the event of an engine failure. It states that systems associated with engine control devices, systems, and instrumentation must be designed to give reasonable assurance that those engine operating limitations that adversely affect engine-rotor structural integrity will not be exceeded in service.

### 3.2.5. FAR 33 - CIVIL STANDARD

United States Federal Air Regulations (FAR) ensure engine durability and are similar to military specifications

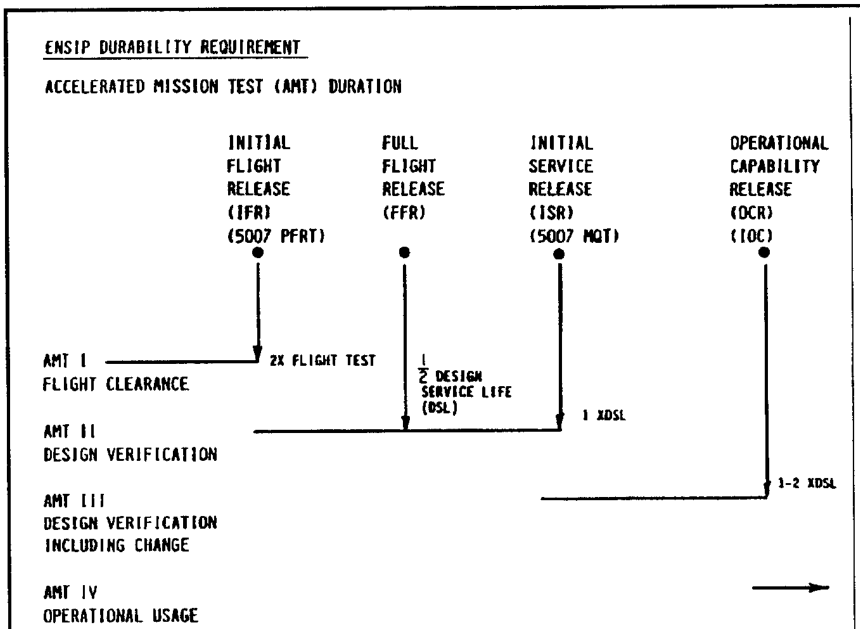


Figure 9 – ENSIP durability requirement

in a number of ways. Requirements include:

- A safe-life practice which must set allowable operational requirements. These must be approved by the Federal Aviation Administration (FAA);
- Components must be classified as to whether they can be contained, or not, in the event of a fracture;
- Components judged non-contained require appropriate life limits and tracking and include disks.
- Components which may cause significant secondary damage are also to be life-limited and tracked and include: shafts, seals, and fan blades;
- Component lives may be set through analytical prediction and/or component testing.
- Initial certification may be to 33% of the predicted safe-life by component testing. Additional testing is permitted to extend the safe-life (life extensions may be up to 1 cycle extension for every 2 cycles tested);
- The Certified Life is an allowed percentage of the analytically predicted safe-life;
- Mission definitions may be defined by the manufacturer;
- The required suitability and durability of materials

is defined;

- The acceptability of the materials test data base must be demonstrated/proven;
- Conformance must be controlled by approved specification;
- The engine design and construction must minimise the development of an unsafe condition of the engine between overhaul periods.

### 3.2.6. UK DEF STAN 00-971

This document has superseded Def Stan 05-2100, which superseded D Eng RD 2100 and 2300. The requirements tend to be harmonised with other standards, but include special requirements based on UK military experience.

### 3.2.7. JAR-E - COMMERCIAL STANDARD

Joint Air Regulations are a harmonisation of existing national regulations into an international requirement. Most nations with aviation design and operating requirements are involved in their development. Without too much loss of accuracy, they may be considered as a combination of the US FAR, MIL-STD-5007E and the British CAR regulations. For civil usage they will replace the BCAR requirements.

### 3.2.8. CANADIAN STANDARDS

In Canada, two documents that relate to airworthiness were produced in 1996. The first is entitled 'Review of the Regulatory Requirements for Aero Engine Design'. It contains a review of the airworthiness certification processes and requirements for aeroengine design changes of Transport Canada (TC), the United States Federal Aviation Authority (FAA), the Department of Defense, the Civil Aviation Authority (CAA) in the UK, and the European Joint Airworthiness Authority (JAA).

The second document, 'Qualification Methodology for Advanced Gas Turbine Repair/Rework', provides gas turbine engine operators with qualification methodologies for airworthiness certification, design and repair and rework changes. The airworthiness certifications are based on requirements of Transport Canada for design approval applications. The document includes Failure Mode, Effects and Criticality Analysis (FMECA) which will enable design authorities to assess component criticality, typical failure modes and failure probability.

### *3.2.9.* OTHER EUROPEAN STANDARDS

Most multinational-European military-aircraft projects create their own airworthiness standards, for both the aircraft and the engines. Examples are Tornado and RB 199, Eurofighter and EJ 200, Alphajet and Larzac.

## 4. THE APPLICATION OF STANDARDS

The transport, fighter and helicopter applications are each influenced by durability requirements that are placed on them by the government or civil entity that is procuring them. It is not unusual for further requirements to be placed on a particular gas turbine design, if it is purchased by an agency that was not involved in the original procurement or certification process. A good example of this is the imposition of damage tolerance requirements on military transport engines that were originally designed in accordance with safe-life practices.

Transport engines generally start their design history under civil regulations which may include (FAR or JAR-E). As damage-tolerance standards have become necessary for United States Air Force applications MIL-STD 1783 has been applied to these engines. Adherence to this standard may lead to the application of additional testing, analysis, and inspection requirements. In some cases, limited component redesign may also be necessary to meet the new design safety requirements.

Fighter engines are not often designed to civil requirements, but can include both safe-life and damage tolerance design approaches, as encompassed in the MIL-E-5007E specification and MIL-STD-1783 standard respectively. It should be noted that the United States Air Force and Navy requirements are different. The Air Force practices the damage tolerance philosophy and the Navy practices the safe-life approach.

Since 1995, the US Navy has introduced refinements to the MIL-E-5007E practices that include damage tolerance analysis. In this case, fracture critical components are designed to LCF and damage tolerance, while managed to crack initiation.

Helicopter gas turbine engines are procured for United States forces predominantly by the Army and Navy. Safe-life practices are followed for these applications with representative specifications including MIL-E-8593A for turbo-shaft engines. On the civil side both FAR and JAR-E specifications are applied to helicopter engines.

## 5. FIGHTER ENGINE APPLICATIONS

The earliest aero gas-turbine engines were designed for fighter applications. Prior to the introduction of military-engine durability standards, engine life was determined by testing. Initially, lives were so short that flight duration was limited by the failure of turbine blades, and examples from this period can be seen in the Deutches Museen (Museum) in Munich. Within a few years gas turbines had developed to the point, where lifing was based on engine-starts and flying hours, but frequent inspections were still required. Major improvements in engine life began to appear in the 1950s and 1960s with the advent of new materials, and improved engine designs. The following control-system comments pertain to engine usage and monitoring systems.

### 5.1. HYDRO-MECHANICAL CONTROLS

### 5.1.1. FRANCE

#### *5.1.1.1. ATAR*

ATAR is a jet-engine family developed since the 50's and during the 60's. The different models equipped the Mirage 3, 4, V and F1 family, the Etendard and the Super Etendard, the Cheetah and the Panthera. The early models were controlled by a hydro-mechanical system; the latest mark (9K50), which powers the Mirage F1 and which entered service in 1969 is controlled by a hybrid analogue computer and hydro-mechanical system.

Initially the mechanical loads were limited by over-speed

resistance and long-term creep elongation. The life limit was fixed in Engine Flying Hours (EFH). Cyclic limits were determined by engine ageing testing and observations made on operational engines. For the newest marks a complete verification of LCF stress was undertaken and used current numerical methods correlated with photo-elastic models.

#### *5.1.1.2. LARZAC*

The 04-C5 Larzac engine entered service in 1979 to power the Alpha Jet. In 1984 the 04-C20 version with higher thrust and temperature was developed. Both versions use a hydro-mechanical regulation driven by an analogue computer, and were designed to safe-life standards. In mid-85, some engines of the French Air Force were equipped with a tape recorder to monitor flight parameters. These parameters were replayed in a ground station to calculate the life consumption associated with each recorded flight. This allowed the correlation of all flight profiles with their damage counts and the determination of a cyclic exchange rate between engine flying hours and the different flight profile categories. The cyclic exchange rates for other operators were determined using their theoretical mission profiles. Life extensions are authorised after the evaluation of disc residual life performed on normally serviced parts using spin pit tests.

### 5.1.2. UK

The Pegasus engine entered UK operational service in 1969 in the Harrier GR1 and in the US in 1972 in the AV8A (Mk 402). The basic engine design dates from the late 1950's. It is based on safe-life procedures. This has not changed through subsequent up-rated engines. The lives of critical parts are defined from specimen and component testing using a 1 in 750 probability of detection of a 0.38-mm crack length with 95% confidence. The lives in operational service are controlled in terms of flight hours using cycles-to-hours exchange rates. These exchange rates for the United Kingdom Services are based on measurements from a small sample of operational aircraft using the EUMS equipment and for other operators on assumed mission profiles. The Engine Usage Monitoring System (EUMS) records the basic engine parameters during normal flying and these recordings are replayed in a ground station to analyse the LCF cycles consumed.

The Adour engine entered service in 1974 in the Jaguar aircraft and critical parts lives were controlled in the same way as with the Pegasus. In 1976, the engine entered service in the Hawk aircraft. Since 1980, it has been used by the Red Arrows display team. The display flying carried out by the Red Arrows was known to generate a wide variety of engine usage rates (a factor of 40 between extremes). In order to mange this a simple low cycle fatigue usage counter was fitted to the Red Arrow Hawk aircraft and has been supplied to several other fleets. An updated version of this unit is currently being applied to the whole RAF Hawk fleet. The retrospective application of a usage monitor to a 'pre-electronic' engine is relatively difficult. This is because aircraft wiring changes are needed to get the signals (speed, temperature, etc.) to an appropriate place in the airframe, where a monitoring unit can be fitted.

### 5.1.3. US

#### *5.1.3.1. AIR FORCE*

The TF30 engine entered production in the mid-60's, on the F111 and A-7P aircraft (See Portugal in App 2) when durability standards were handled by engine test and any durability requirements were included within the engine specification (Taylor, 1969). The TF-41 engine also entered service at the same time on the A-7 aircraft. As shown in figure 2 lessons learned from the TF30 program were used to help develop MIL-E-5007. Life extension, and a retirement for cause evaluation on the TF30 have required that a damage tolerance analysis, a focused inspection, and refurbishment or modification of components be conducted to extend the life of critical components. Under the life extension plan, the components are allowed to operate for one inspection interval following the refurbishment prior to being replaced or retired.

The F100-PW-100 was in production from 1976 to 1986 and used on the F-15 fighter. This engine was developed using safe-life procedures and was the subject of a structural assessment, conducted from 1978 through 1982, prior to the development of the ENSIP (MIL-STD-1783) specification. This study sought to determine the impact of applying damage tolerant procedures to an engine that had been designed using safe-life procedures, and what would be involved in implementing such a design approach on an existing engine.

#### *5.1.3.2. NAVY*

The J52 engine, entered operational service in 1965 in the A6 (J52-P-6) aircraft and made its first flight on an A4E (J52-P-6A) aircraft in 1961 (Taylor, 1969). However, the J52 started out its life as an air-to-surface cruise-missile engine where durability was not a concern. The engine was subsequently updated to the safe-life process and over 2,600 engines were produced. The J52 uses a hydro-mechanical engine control.

The TF30 entered service on the F-14 fighter in 1972. It was developed using safe-life procedures that included the MIL-E-5007 standard. The TF30 uses a hydro-mechanical engine control. After several years of operational service, TF30 rotors were found to have LCF cracks, well before the design life requirement was reached. In an effort to keep the fleet flying, whilst a redesigned disk could be qualified, component lives were re-analysed using damage tolerance procedures. (1/1000 initiation + life to rupture from a 1/32-in crack – approx.0.8-mm - on a 1/1000 basis) Components were inspected at overhaul with normal inspection procedures (FPI), returned to service and flown to one half of the crack propagation life. By this time, the redesigned disk was qualified and began to replace the high flight-time components in the field.

This is a good example of the use of damage tolerance analyses and field inspections to mitigate risk until a redesign could be deployed.

## 5.2. Hybrid Control Examples

### 5.2.1. France

#### 5.2.1.1. M53-5

The M53-5 entered service in 1978 to power the Mirage 2000. The engine control system is composed of an analogue full-authority regulator with redundancy. A hydro-mechanical system that covers the full flight envelope is provided for the safety mode. The engine has a modular design. Life is determined using safe-life procedures. The main damaging parameters are fatigue and creep. Individual component damage-tracking, based on the flight time, is performed on each module.

## 5.3. Digital Control Examples

### 5.3.1. France

#### 5.3.1.1. M53P2

The M53-P2 entered service in 1985, with a full-authority digital control Engine Control System (ECS). The engines operated by the French Air Force since 1987 have been individually equipped with an Engine Life Monitoring System which calculates the fatigue and the creep damage and stores the residual life for about 20 critical part locations. After engine initial build or rebuild the on-board system is loaded with the engine configuration i.e. serial numbers and residual life of the monitored components. If the component is new, the residual life is the initial life. The life calculation is performed in real-time by a dedicated system using the flight parameters transmitted by the engine control system. Flight-by-flight life consumption can be displayed to the flight mechanics. But most of the time it appears that a periodic, typically once a week, downloading of the life consumption to the on-ground component life database is adequate. It allows a precise enough monitoring time-step while reducing maintenance workload. Since that time the ELMS has evolved several times because of the acquired experience of the engine behaviour in operation.

The fatigue life and creep lives, and the inspection schedule of the monitored components are based on limits expressed in a specific flight-damage unit consistent with the on-board damage calculation. In the 1990's the on-board ELMS was introduced as a standard engine component and is now used by several customers.

### 5.3.2. UK

The Pegasus engine entered operational service in 1969 in the Harrier GR1 and in 1972 in the AV8A (Mk 402). The current Pegasus Mk 408 has a digital electronic control system. The engine design is based on safe-life procedures. The lives in operational service are controlled in terms of flight hours using cycles-to-hours exchange rates. These exchange rates for the UK Services are based on measurements from a small sample of operational aircraft using the EUMS equipment and for other operators on assumed mission profiles.

Early aircraft were fitted on a sample basis with the Engine Usage Monitoring System (EUMS) which recorded the basic engine parameters during normal flying and these recordings were replayed in a ground station to analyse the LCF cycles consumed.

With the introduction of the Harrier GR Mk5 the Engine Monitoring System was introduced. This was fitted on a fleet wide basis and computed life usage on selected components in near real-time. At the end of each flight the results were downloaded and transferred to the Harrier Information Monitoring System (HIMS). This provided fleet life management tools.

### 5.3.3. US

The introduction of electronic engine controls in gas turbine engines coincided with the application of damage-tolerance design practices in fighter applications for the United States Air Force. The F100-PW-220 turbofan was one of the first engines designed to the MIL-STD-1783 damage tolerance standard, and was introduced into operational service in 1986.

The F100-PW-229 turbofan is a higher thrust derivative engine that followed the F100-PW-220 program, and was designed to meet damage tolerance standards. It contains a second-generation digital electronic engine control, and is used in F-15 and F-16 aircraft as well.

The F110-GE-129 includes a single channel Digital Engine Control (DEC) with a hydro-mechanical backup. The F110-GE-129 was designed in accordance with damage tolerance standards. Used in the F16C and D fighters, this engine was introduced in the late 80's.

## 6. Transport Engine Applications

### 6.1.1. Hydro-mechanical Controls

#### 6.1.1.1. US

As with fighter engines, prior to the introduction of military standards and specifications on engine durability, engine testing to determine life was the rule.

One of the earliest US gas turbine engines was the J57 on the B52 bomber (Models C, D, F, G) and the KC-135A Tanker. This engine was introduced in early the 50's and is similar to the JT3 engine used on commercial Boeing 707 models (Taylor, 1969). This engine uses mechanical controls.

The TF33 has replaced the J57 on the B52 bomber (Model H) and is also used on the Lockheed C-141 B, Boeing E-3A, KC-135E and C-18A

### 6.1.2. Digital Control Applications

#### 6.1.2.1. US

Three US civil gas turbine engines that were originally designed to safe-life standards (i.e. FAR) have since been assessed to military damage tolerant standards (i.e. MIL-STD-1783). The assessment and subsequent actions to meet the damage tolerance standards do involve additional work and cost but can pay benefits later in terms of life cycle cost. The engines that have been scrutinised with damage tolerance standards are:

- The JT15D-5B turbofan for the Jayhawk T1A;
- The F117-PW-100 turbofan for the C-17 transport;

Each of these engines features electronic controls, which have been standard from the start on the PW2000 series of engines from which the F117-PW-100 turbofan is derived. The electronic controls were later added to the JT15D-5B engine, which began its derivative development from the original JT15D-1 engine in the late 80's. These engines range in capability from 3,000 to 60,000 pounds of thrust which shows that damage tolerance procedures can be applied to any gas turbine application regardless of size. Each program had to:

- Identify critical components;
- Conduct thermal and stress analyses;
- Perform durability and damage tolerance analyses;
- Characterise materials for crack growth behaviour;
- Qualify inspection capabilities.

In the case of the JT15-D Accelerated Mission Testing (AMT) was conducted to a cycle that was very different from the commercial cycle for which it was originally designed. Additionally, the following tasks were required:

- Tracking to update the operational mission definition is planned;
- A life management plan had to be developed;
- A plan to modify hardware, if necessary, had to be put in place;
- Additional inspections, over and above those required for commercial applications, could be called for.

## 7. HELICOPTER ENGINE APPLICATIONS

### 7.1. HYDRO-MECHANICAL CONTROL EXAMPLES

Like transport aircraft engines, helicopter engines in this category were generally designed to safe-life standards.

#### 7.1.1. FRANCE

The TURBOMECA ARRIEL 1 turbo-shaft engine was developed in the 70's, entered in service in the early 80's and is installed on the AS365N2, AS350B2, S76c, and Agusta A109K helicopters. It uses a hydro-mechanical engine control. The in-service low-cycle fatigue counting is carried out according to either the 'lump' or the 'recommended' methods. The latter is based on the counting and level of complete and partial cycles. There is some benefit using the recommended method because it counts fewer cycles than the 'lump' method.

An electronic device is proposed as an aid to cycle counting. This aid uses an algorithm based on the recommended method, with rainflow decomposition of the engine speed cycles. The rainflow technique identifies and pairs the peaks and troughs of the engine rotor speeds into cycles. The damage is then computed for each cycle and added to the cumulative total.

#### 7.1.2. UK

The Rolls-Royce Gem engine entered service in 1976 in the Westland Lynx helicopter and used a hydro-mechanical control system. The critical parts were designed using safe-life procedures with the overall life-control process similar to that used for the Pegasus.

#### 7.1.3. US

Engines are designed to safe-life standards, (FAR 29, 33, and JAR-E) and MIL -E-8593A (Aircraft Engine Turbo-shaft/Turboprop). The Army and Navy are the primary purchasing agencies and set the requirements.

All of the turbo-shaft engines currently in the United States Navy inventory completed design and qualification prior to 1985. They were designed using safe-life procedures. If, in the future, component life limits are reviewed and updated for these engines, damage tolerance may be assessed. Two points of caution should be considered. First, the engine components were not originally designed with damage tolerance in mind and the likelihood of usable crack growth life existing is considered small. Second, it may be more cost effective to simply replace components with a new (zero time) part than to conduct periodic inspections.

### 7.2. HYBRID CONTROL EXAMPLES

#### 7.2.1. FRANCE

The MAKILA turbo-shaft engine is installed on the SUPER PUMA helicopter. This engine was designed using safe-life procedures. It uses a hydro-mechanical engine control unit for the generator and an electronic engine control unit for the power shaft. The LCF counting philosophy is the same as for the ARRIEL1.

### 7.3. DIGITAL CONTROL APPLICATIONS

#### 7.3.1. FRANCE

##### *7.3.1.1. ARRIEL2*

The TURBOMECA ARRIEL2 turbo-shaft engine powers the S76C+, AS365N3, and AS350B3 helicopters, and is equipped with a single channel FADEC. Certification is in accordance with the JAR-E regulation and its design is based on a safe-life analysis. The initially authorised in-service life limits of the rotating parts are to be progressively extended following a life management plan. The in-service low-cycle fatigue counting is carried out according to one of the following methods:

- The 'lump' method;
- The recommended method based on partial cycle counting.

The recommended method is programmed into the Electronic Engine Control Unit. It uses rainflow decomposition and is the primary counting method.

The ARRIUS2 engine design and control philosophy is the same as that for the ARRIEL2. In some applications, the use of emergency rating (One Engine Inoperative) is also electronically counted.

#### 7.3.2. INTERNATIONAL

##### *7.3.2.1. FRANCE AND UK*

The Rolls-Royce Turbomeca RTM322 turbo-shaft engine was developed at the beginning of the 80's. Engine version 01/1 was certified in accordance with the JAR-E

regulations. It is also qualified in accordance with the UK DEF STAN 00-971 for the EH101 and MERLIN helicopter applications. The RTM322 is to be mounted on the NH90 (JAR-E) and APACHE (DEFSTAN 00-971) helicopters. It includes a FADEC with a twin channel electronic engine control. Based on the safe-life concept, the low cycle fatigue life counting for the rotating parts is calculated by the helicopter control unit using a rainflow algorithm.

### 7.3.3. UK

Variants of the Rolls Royce Gem engine (from 1984) in the Lynx, WG 30 and Augusta A129 aircraft were fitted with a digital electronic control system. The critical parts were designed using safe-life procedures.

The Augusta A129 helicopter incorporates a central on-board computer system that carries out several avionics functions. Since all of the necessary signals are available in suitable electronic form, this central computer is used to carry out the engine usage monitoring function.

In the new JAR OPS regulation, single or twin engine, class 2 helicopters are no longer allowed to take off in hostile areas. In some countries, exceptions may temporarily be accepted provided that precautions, including automatic cycle counting, are taken to keep the probability of in-flight shut down during the exposure time lower than a specified value.

## 8. SIGNIFICANT ISSUES OF COST AND SAFETY

The transport, fighter and helicopter applications are each influenced by durability requirements. The impact of different durability standards is discussed elsewhere in terms of the following.

- Parts classification procedures;
- The on-condition maintenance approach versus a more intensive inspection and preventative maintenance.

### 8.1. PARTS CLASSIFICATION DISTINCTIONS

Varying levels of classification exist between specifications, which include:

- (ENSIP) which produces five categories of parts (safety, mission, durability, durability non-critical, other) depending on the impact to the aircraft and its performance, and cost.
- MIL-STD-5007E which considers rotating components and casings. These parts are further differentiated as hot parts, cold parts and critical parts.
- FAR 33 requires that all parts, such as disks, that would be uncontained in the event of a fracture have life limits published in a table of limits that is approved by the FAA. Parts, such as shafts and seals that would be contained but would cause significant secondary damage also have life limits.

These classifications have worked successfully for their intended applications. A failure mode effects and criticality analysis (FMECA) is used to classify components for the ENSIP process but the procedure for the FMECA is actually described in MIL-STD-1629. FMECA looks at whether or not a component failure will jeopardise safety or completion of the mission, and accumulates the risk of failure at the component level so that it can be evaluated on a system basis. The effect of the more restrictive classification procedure in ENSIP varies from application to application, with single versus multiple engine applications showing the most striking differences. An example of the level of components classified as fracture critical is 59% for the single engine application against 21% for a similar multiple engine application. Interestingly, the rotating components and cases named in FAR 33 and 5007E will be found as a subset of the MIL-STD-1783 classification and will be considered fracture critical.

### 8.2. ECONOMIC LIFE DECISIONS ON A COMPONENT OR MAJOR STRUCTURES BASIS

Under the ENSIP philosophy a classification is forced on every component. This places it into one of the categories described earlier. A fixed period of unrepaired service, either safety limit or inspection interval, for safety and mission critical components is required. Special procurement and tracking procedures are applied according to the durability-critical classification level of each component. This process forces appropriate maintenance actions down to component level. Flight safety and mission capability are the paramount concerns, while cost is emphasised to a lesser degree in this approach to maintenance actions.

For those following safe-life procedures, components are not classified in as much detail. When maintenance is required, all parts are not necessarily disassembled and inspected. In this case, inspections are normally conducted only when the aircraft, engine, or component is available for disassembly. Components are only replaced once a minimum fatigue life is reached and an overhaul required. Components can also be classified with different life limits depending on whether they are in the hot or cold section of the engine. This influences the period at which a component or engine module is brought in for overhaul and inspection. Frequently, the hot section has half the life-between-overhaul capability requirement of the cold section. While flight safety is the paramount concern, cost is also an important consideration. Because inspection facilities are not required in the infrastructure as part of the lifing policy (unlike ENSIP), their non-availability may influence maintenance decisions that directly affect safety. For example, a lack of fleet management tools may lead to optimistic decision making because adequate information is not available.

## 9. SUMMARY

Design practices that emphasise durability and the advent of electronic controls have improved the reliability of helicopter, transport, and fighter gas turbine engines.

Developed as a response to engine structural problems, damage tolerance and safe-life practices are part of wider specifications to ensure engine durability. The advent of

these improved design methods coincides with the introduction of electronic engine controls, which have contributed to improved engine reliability and safety, by providing better data to usage monitoring systems. There has been a continuing effort, since durability problems were recognised, to improve engine design, test, and maintenance activities.

Safe-life design procedures practised by commercial engine manufacturers parallel those of the military but the associated maintenance procedures can differ greatly. Generally, when a gas turbine design is originally developed for the commercial sector and later adopted for military use, it requires:

- Additional analysis;
- Modified maintenance practices;
- More focused inspection requirements;
- Mission assessment and testing;
- Redesign to improve durability.

These requirements are especially true when changing from a safe-life to a damage tolerance approach. Damage tolerance practices do offer advantages over safe-life methods in terms of life cycle cost but may be difficult to implement by some military services because of the logistics involved in both facilities and inspection equipment.

## 10. CONCLUSIONS

Improved durability design practices and the introduction of electronic engine controls to gas turbine engines coincide historically. 'Safe-life' and 'Damage Tolerance' design practices are the most commonly followed procedures to ensure reliable engine design. Either method can be chosen by the agency procuring the design. The decision is usually based on a number of factors such as:

- Logistics;
- Infrastructure investments;
- Common operational procedures;
- Common maintenance procedures;
- Common training;
- Contracting requirements;
- Recent experience.

Based on the number of engines reviewed, the safe-life method is more widely practised than the damage tolerance method. However, the competitive nature of the engine market place is driving all new engine designs towards a damage tolerance approach for highly stressed components.

Comparison of the safe-life and ENSIP based processes shows that each offers advantages and disadvantages to the services that practice them. The advantages of a safe-life practice are:

- The emphasis on the least maintenance possible;
- Maximisation of 'on wing' time without inspection;
- A decrease in the facilities and equipment needed to conduct maintenance inspection actions.

Disadvantages of the safe-life method include:

- Fewer fleet management options;
- No focused inspection process available if cracking problems develop;
- The premature retirement of 999 out of 1000 components.
- The need for readily available replacement parts.

Modifications to the safe-life practice offer opportunities to extend the life of components through fracture mechanics, after a minimum safe-life is reached. Unfortunately, the margin is generally smaller than for a component that is designed to the damage tolerance standard. Damage tolerance design practices have these advantages:

- Increased use of parts beyond LCF limits may be permitted, subject to regular inspection;
- More thorough testing and verification of designs;
- It provides a more readily available, focused inspection capability to address cracking problems.

The disadvantages of imposing damage tolerance practices include:

- It is more costly to implement than the safe-life process;
- It adds weight to the design;
- It requires a larger infrastructure;
- Parts' handling is increased
- It may dictate the only viable management system for introduction of mature fleets.

Commercial engine designs have been certified to a military ENSIP standard but this practice carries extra cost, such as:

- Additional analysis;
- Modified maintenance practices;
- More focused inspection requirements;
- Mission assessment and testing;
- Redesign.

## 11. RECOMMENDATIONS

Neither safe-life nor damage tolerance, is right for every agency on every occasion. When choosing a design practice, consider the following:

- Fleet utilisation;
- Access to inspection facilities;
- Life cycle cost comparison (i.e. safe-life versus damage tolerance approaches);
- Choose the best approach based on a valid comparison.

## 12. REFERENCES

Kuhlberg, JF, Bartle, RG, Glomb, WL, *FADEC: A Continuing Technology Success Story*, The Australian Aeronautical Conference, Melbourne 9-11, October 1989.

Ogg, JS, *Engine Durability & Damage Tolerance Assessments*, Proceedings of the 1987 Aircraft/Engine

Structural Integrity Program (ASIP/ENSIP) Conference.

MIL-E-005007E, *Military Specification for Engines, Aircraft, Turbojet and Turbofan, General Specification for*, Prepared by United States Navel Propulsion Center, 1 September 1983.

MIL-STD-1783, *Engine Structural Integrity Program (ENSIP)*, Prepared for the United States Air Force by Aerospace Industries Associates of America, Inc, 30 November 1984.

Taylor, JWR, *Combat Aircraft of the World from 1909 to Present*, GP Putnam's Sons, 200 Madison Avenue New York, NY 10016, 1969, pp 458, 496-498.

# Chapter 3
## Maintenance Policies and Procedures

by

*(O. Davenport)*

# 1. INTRODUCTION

There is a multitude of maintenance policies and procedures that may apply to the management of fracture critical rotating parts in aero gas-turbine engines.

Maintenance policy is influenced by:

- The era in which the engine was developed;
- The sophistication of the aircraft in which the engine is installed;
- The information technology base of the operator;
- Design analysis improvements by the engine manufacturer;
- The influence that the operator has, with the aircraft and engine manufacturers;
- The inherent life of the engine components.

There does not appear to be a single solution to this most complex situation and it is incumbent on the manufacturer, airworthiness authority and operator to select appropriately from the broad array of possible solutions. The challenge is difficult in an era of declining defence budgets and the dramatic expansion of information and sensor technology. Still, this decision will significantly affect the cost, safety, and readiness attributes of the weapon system.

There are three primary criteria for the success of a maintenance policy for fracture critical rotating engine components:

- It must ensure the airworthiness of the aero engine by ensuring that the parts are removed from service in time to prevent catastrophic failures;
- It must be affordable;
- The engines must provide the combat readiness needed to support the national security strategy.

The first is clearly achieved when there are no catastrophic mishaps that are caused by this class of components. The second criterion is much more obscure. How does one know when the lowest maintenance costs have been achieved? Because of its wide influence and the repercussions of any serious shortcomings combat readiness, in many cases, overrides the drain on the national treasuries. This means that affordability is often expressed in the availability of resources to purchase, stock, and issue replacement parts and components. When the policies become unaffordable, readiness and sometimes airworthiness may be degraded.

Older systems are generally maintained in accordance with one of the following policies:

- Fixed hourly or cyclic interval;
- On-condition through management of turbine temperature margin either calculated or measured;
- A more complex system of data recording and analysis.

Newer systems may use any of the above and additionally may include damage tolerance techniques for lifing or life extension.

## 1.1. ENGINE CLASSIFICATIONS

For our purposes, engines may be classified into three main groups: Fighter, Transport and Helicopter. These have different operating regimes and weight sensitivities. Frequently, the greater the potential gains from a life-usage monitoring system may be the greater the difficulty in finding the weight budget, and the space. For psychological reasons, the weight sensitivities sometimes increase dramatically when additional weight in the form of a life monitoring system is proposed. Because the operational requirements for similar aircraft types also vary markedly between operators there are likely to be small but significant differences between the maintenance policies of different operators with the same aircraft and engines. An example of this diversity may be found in the European Tornado aircraft. The UK records data on a very small sample of aircraft, Germany monitors the whole fleet, and Italy is changing from no monitoring to using the German OLMOS system.

### 1.1.1. FIGHTER ENGINES

Today's fighter engines have undergone remarkable improvements in the operability of the engine across the flight regime. Most aircraft now have unrestricted throttle movement with stall-free operation at all points of the flight envelope as a standard feature. They also have slam-acceleration times of a few seconds, rather than the 30 to 60 seconds of the earliest operational gas turbines. This results in the potential accumulation of many types of cycles resulting from throttle changes. For many current engines usage is measured parametrically on an equivalent or reference cycle basis. For USA engines a common parameter is "Total Accumulated Cycles" (TACs). These may relate to throttle movements or spool speed, and are defined in Appendix 1. Some modern engines also limit the effect of rotor transients by modulating thrust using the variable vanes, exhaust nozzle position and fuel flow to temporarily maintain a constant rotor speed. While still in use in many NATO countries, older engines have operating limitations that may limit throttle movements. Therefore, these engines do not experience the same myriad variety of cycles and cycle types as the more modern engines.

### 1.1.2. TRANSPORT ENGINES

Turbine engine powered transport aircraft may have either turboprop or turbofan/turbojet engines. In general, the age of the system determines the engine maintenance policy. Transport aircraft tend to fly with far fewer throttle movements than fighter aircraft. However, this is not always true. The following cases are possible exceptions:

- When automatic landing systems are in use a large number of throttle excursions may be used;
- Tanking (in-flight re-fuelling) manoeuvres;
- Low level freight drops.

Turboprop engines are typically designed to be maintained either on-condition or at a fixed hourly interval. Typically, on-board condition monitoring and life measurement equipment is not installed in transport aircraft, although the UK VC10 and TriStar fleets are

notable exceptions.

### 1.1.3. HELICOPTER ENGINES

Helicopters have been fitted with turbo-shaft engines for many years. Indeed, the installation of gas turbines into helicopters transformed their military effectiveness in the 1960s. Again, the age of the system and the design requirements determine the engine maintenance policy. Turbo-shaft engines are typically designed with cyclic-life limited components and are maintained to a hard-time hourly limit. Numerous rotating components are life limited due to the rotor-speed related stresses that they endure. Power turbines in turbo-shaft engines are effectively life limited by thermal cycling because they operate at a constant rotor speed for power settings above idle. However the power turbine disc and shaft still experience one major LCF stress cycle per flight, and torque levels vary frequently. Special considerations for helicopter engines are:

- Torque-matching of engine power outputs to protect helicopter gearboxes;
- Multiple engine control system adjustment to ensure that one engine does not use excessive life, compared to the others;
- The 'automatic' engine throttle movements resulting from Power Turbine Governor system operation, to keep the helicopter rotor speed constant despite collective and cyclic pitch inputs from the pilot.

## 2. COMMERCIAL AND MILITARY DIFFERENCES

The operational mission that commercial engines are subjected to is much less complicated than the military cycle. This difference is best illustrated by the operational cycle of the military F117-PW-100 engine, contrasted with that of its commercial cousin, the PW2037 engine. The F117 experiences one or more major cycles with each practice touch and go, cycles due to airdrops, cycles due to use of thrust reversers for backing and turning, etc. The cyclic accumulation rate is about 2-3 times the PW2037 on a cycle/hour basis. Clearly, adapting a civil engine to a military application can require design modifications, increased inspection requirements, and alternative maintenance approaches. An exception to the above is that engines on regional jet transports accumulate cycles faster than similar engines in long range endurance such as reconnaissance or maritime patrol applications.

Commercial operators tend to see to the care of their own engines once they are purchased from a manufacturer. Therefore, the overhaul and repair facility capabilities, and the quality of the approach to maintenance vary greatly. Usage of the engines and aircraft also vary as to the geographical region served and utilisation (short versus long flights). Government regulations do enforce a degree of uniformity in maintaining engines and their safe operation but each carrier is a separate entity, and responsible for its own actions.

In the use of military engines and their maintenance by different countries, there are some similarities to the commercial situation. Again differences in maintenance practices can vary, and here it is up to the manufacturers and the countries to which the engines are sold to arrive at the best process. Exchange of information in the form of maintenance orders from the parent country in which the engine was designed and manufactured and the country that purchased it are the rule.

Usage of the aircraft is another variable. Countries that are smaller geographically tend to have more severe missions (more low level activity with heavy weapons loads and little or no cruise time to engagement). Thus, reduced durability and an increased frequency of maintenance action may result from introducing the engine into a harsher environment than the one for which it was designed.

## 3. MAINTENANCE POLICY & SUPPORT COSTS

The general relationship between the maintenance strategy and the operating and support costs is simple.

- The acquisition cost of the system accounts for approximately half the life cycle cost;
- The maintenance and repair costs account for the remainder.

Many factors affect costs both directly and indirectly. Design choices directly affect the acquisition costs, and are a significant factor in the maintenance and repair costs, especially when viewed on a cost per flight hour basis. The inherent reliability of a design is demonstrated by the initial capability of the machine. Maintenance practices and policies affect this reliability once maintenance actions begin. If the performance is not restored during maintenance, lower reliability and higher operational costs are the inevitable result. If the performance is improved following a maintenance action, then this is almost certain to be due to the correction of a deficiency in the original design.

### 3.1. FIELD STUDIES

There appears to be an optimal relationship between the material replaced at each shop visit and operating and support costs. Commercial experience with CF6-80A engines on the A310 aircraft has conclusively shown that operating and maintenance costs are optimised by minimising the cost per flying hour, rather than shop visit and material replacement costs. Figure 1 shows a typical relationship of material cost per shop visit versus cost per engine flight hour. This was achieved in practice by synchronising maintenance so that a complete engine with accessories is placed in service as a unit. This practice, rather than making module changes on the wing, has produced dramatic reductions in operating and maintenance costs.

This finding is of immense importance to all operators, and efforts should be made to determine whether a similar relationship exists for all fleets. Operators with older engines that are changed as complete Engine Change Units (ECU) will already be following this practice, and should thoroughly investigate the real cost of changing to a modular engine philosophy, should this be proposed.

These benefits are derived from a harmonised maintenance schedule that incorporates both life limited components and performance limited components that are typically in the engine core i.e. compressor and turbine flow path components. Figure 2 shows the potential for engine removals when the maintenance is not synchronised. It is easy to observe from figure 2 that a maintenance program that aligns the module maintenance requirements - life limits or performance - offers the greatest potential for reducing operating and support costs.

### 3.1.1. RAF TRISTAR – 500

The RAF acquired six Lockheed Tristar –500 aircraft in 1984. These had a usage monitoring facility fitted. This system is much more capable than a dedicated engine component usage monitoring system. Benefits reported from the system, after extensive analysis of several years service were classified as economic, and safety.

Economic benefits accrued from the following.

- Avoidance of secondary damage, due to detection of incipient engine failure;
- Avoidance of unnecessary removals;
- Avoidance of maintenance actions and service disruptions;
- Fuel savings due to improved variable inlet guide vane adjustment;
- Planning of engine changes to occur at base.

A conservative estimate of savings, that excluded any manpower considerations, showed a reduction of £72 per aircraft flying hour or £24 per engine operating hour at 1984 prices.

Safety benefits accrued from the following.

- Preservation and protection of the margins in the operational envelope of the Tristar;
- Preservation and protection of the margins in the structural lifing of the Tristar;
- Preservation of the integrity of the structure from undetected damage due to operational incidents;
- Protection of the Controllerate of Aircraft (CA) Release fatigue usage assumptions.

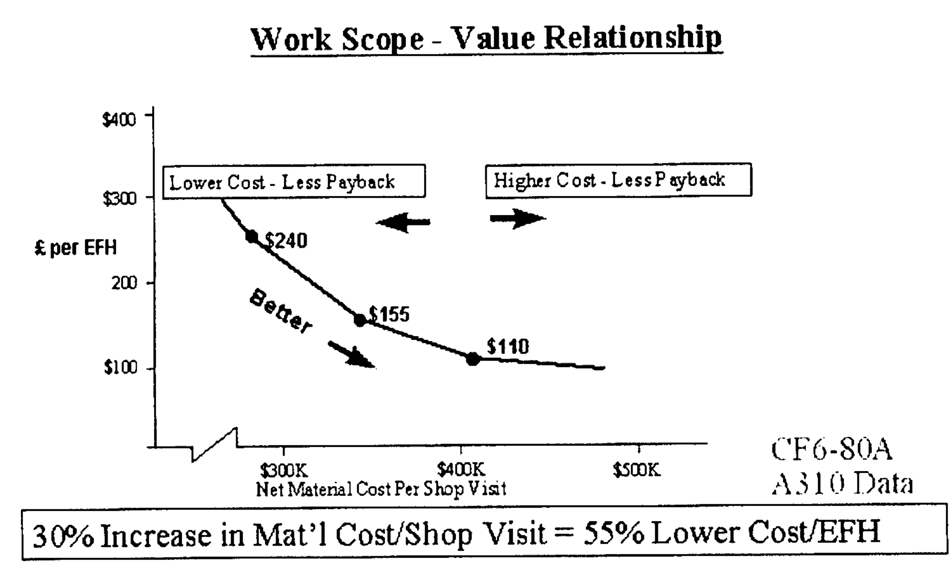



Figure 1 – Optimising engine flying hours cost

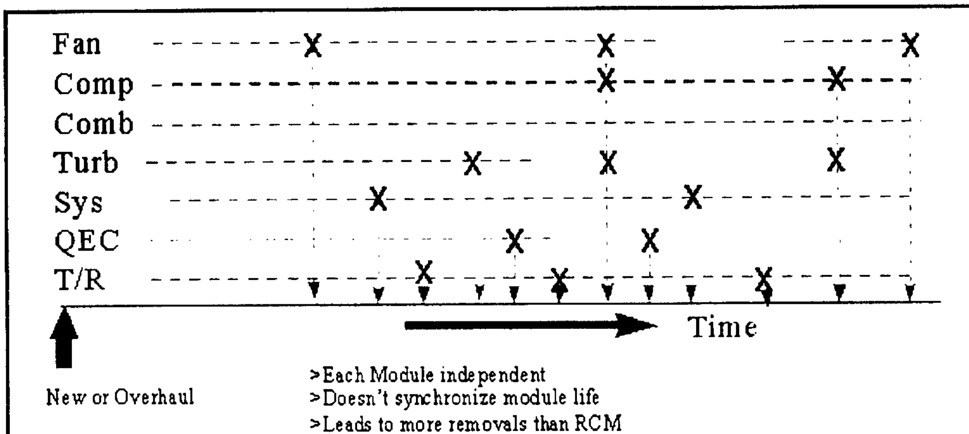



Figure 2 – Synchronising module and engine lives

When multiplied to represent potential savings in a large fleet these savings represent many million pounds per year.

### 3.1.2. UK ENGINE MONITORING RETROFIT WORKING GROUP

An extensive study report by the UK Engine Monitoring Retrofit Working Group on the benefits of fitting engine life counters included the following conclusions:

- Lifing practice was investigated and it was concluded that for future implementations Thermal Transient methods are the only methods acceptable if full Predicted Safe Cyclic Life (PSCL) is to be achieved.
- The cost benefit studies undertaken indicate that on Hawk aircraft fleetwide monitoring will produce significant cost savings from parts and labour. Management benefits are more difficult to quantify but it is likely that examination of Minimum Issue

Service Life (MISL) policy could give rise to further cost benefits.

The Working Group considers that the fleetwide fit of engine monitoring units is supported on safety grounds alone. The statistical analysis undertaken as part of the Harrier/Pegasus data study indicates that small sample monitoring is inherently inaccurate. Application of this statistical method on Hawk Adour and Tornado RB199 data indicated that considerable factors would need to be applied to average exchange rates derived from small samples. Independently, recent agreements on Adour and RB199 data reflect the uncertainty of the small sample methodology, which has resulted in a 30% worsening of the exchange rates.

### 3.1.3. F-16 C/D WITH F100-PW-220

The result of a specific on-condition-maintenance program that focused on restoring functionality at the right time is shown, in figure 3, for the F100-PW-220. When this engine entered into service the Mean Time Between Repairs (MTBR) approached 800 engine flight hours, and the operation and supports cost were quite low. As the engine aged and the component failure rate increased, maintenance was performed at the module level without regard for the overall engine performance and coherence. Consequently, the operator found that the reliability decreased with a commensurate increase in maintenance costs.

A programme, which incorporates the best practices from the commercial fleet, to manage the life of the engines more effectively is currently underway in the USAF. Engine life management is a high payoff, relatively low cost process that significantly improves both readiness and supportability.

### 3.1.4. PW2000 APPROACH

The PW 2000 modular maintenance approach of the mid 80s, figure 5, was evolved due to commercial customer demands for improvement in time-on-wing, reliability, performance and operating costs. The need for an Engine Management Plan (EMP) which optimised engine-management practices based on service experience and development efforts was established. The EMP embodies a 'whole engine' systems level (RCM) approach to maintenance, which is designed to optimise maintenance costs per EFH. The more recent F117 takes full advantage of this commercial experience as well as previous military lessons learned on the TF39 and other engines. An EMP has been established for the F117, which weighs all pertinent engine data in order to make the best maintenance decision, figure 6. From this plan, a customised work package is established for each F117 returned for overhaul. The F117 EMP also takes into account the current spares level, shop load and economic factors in establishing the detailed build requirements for each engine.

This approach is not limited to new products but is performed for all products in operational use, both licence and own development. The philosophy is applied from the initial product feasibility study or definition up to its retirement; the same process is performed to analyse any design change, life or performance improvement requirement, mission profile variation, support and maintenance concept variation. Figure 7 shows, under the support activity, the recurrences of the Life Support Costs versus Cost of Ownership analysis in the various programme phases.

The above analysis originated the pie chart of figure 8. This represents a product for military application, and highlights that inside the system Life Cycle Cost, the relationship between Acquisition Costs of a System and its Operations and Support Costs is generally one to one.

The other LCC elements, Development and Retirement & Disposal Costs, are subject to various deviations originated by technical and design solutions, material utilised etc, that can largely influence their weight on the Total Life Cycle Cost.

Note: In the figure 8 piechart, acquisition cost includes research & development, production investment, and initial support investment. This is not applicable to commercial products where acquisition costs are market driven and recovered via specific commercial agreements (Flight Hour/ Shop Visit fees etc.)

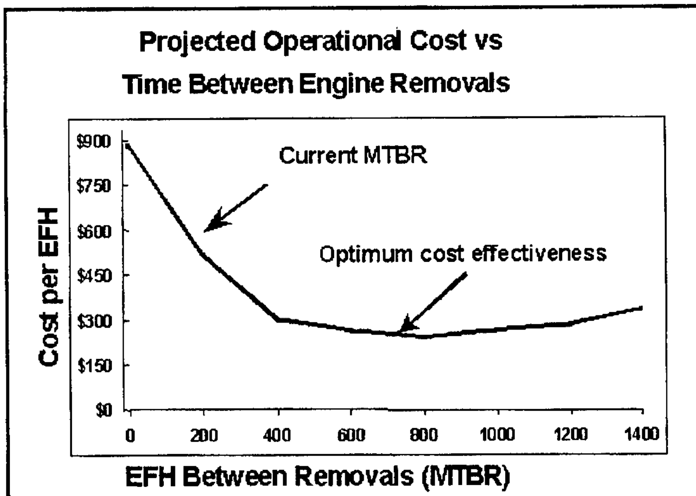


Figure 3 – Minimising Operational Cost for the F100-PW-220

The experiences, gained in the past via several Cost of Ownership Analyses, have highlighted that the number of variables to be considered in the military applications is greater than in commercial ones. In fact, the typical military approach to "System Effectiveness" may drive Product Cost Effectiveness to be "Optimised" instead of "Minimised".

Figure 9 shows the relationships and dependencies between Life Cycle Costs and Systems Effectiveness. An example of the application of this philosophy is shown in Figure 10.

The final aim of the study was to compare the influence on Operations and Support Costs of three different Engine Thrust settings (performed via Engine Control Unit) that were requested by the operator to expand the mission profile flight envelope.

The result is a 'Product Handling Programme' that provides utilisation 'windows' so that the operator can control costs or performance according to current needs.

## 3.2. DAMAGE TOLERANCE COST/BENEFITS

A comparison of the basic differences between MIL-STD-5007E, a predominately safe-life approach, and MIL-STD-1783, a damage tolerance approach is made in chapter 2. Fracture mechanics material characterisation, parts classification, and sub-system damage tolerance verification testing create extra effort in developing the damage tolerant design. These efforts include:

- More refined finite element analysis to define stress states for crack growth analysis;
- Crack growth life computations;
- Crack growth characterisation of materials;
- Verification testing of a few prototype parts;
- Increased inspection costs (focused FPI and eddy current) on each engine;
- Inspection facility acquisition.

As discussed elsewhere, very large safety and operational benefits go along with these increased costs. The primary result being the virtual elimination of LCF failures in the fleet. Not all components are defect free when manufactured. The anomalies can cause premature failures and a safe-life approach does not recognise these defects. Further, if RFC is used for the life management approach a large life cycle cost savings may be achieved over the life of the engine. Operational safety is the primary justification of the ENSIP costs.

## 3.3. RELIABILITY CENTRED MAINTENANCE

The reliability centred maintenance (RCM) concept was first employed in the early 1980's. The philosophy is based on an examination of the operational consequences of failure for each component. If the consequences of failure are unacceptable, a maintenance task is established for the component. The result is a program that achieves safety and operational goals while minimising costs, maintenance man-hours and material usage. RCM uses engineering decision logic to evaluate each failure mode and

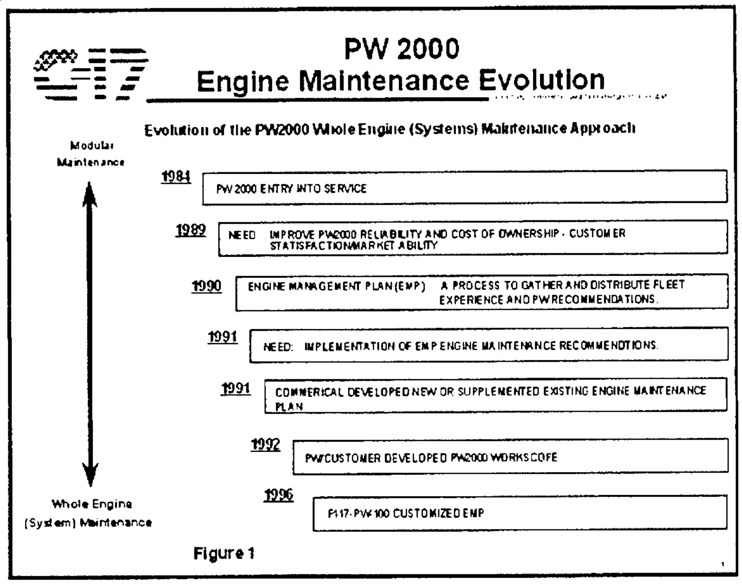



Figure 4 – Modular maintenance example

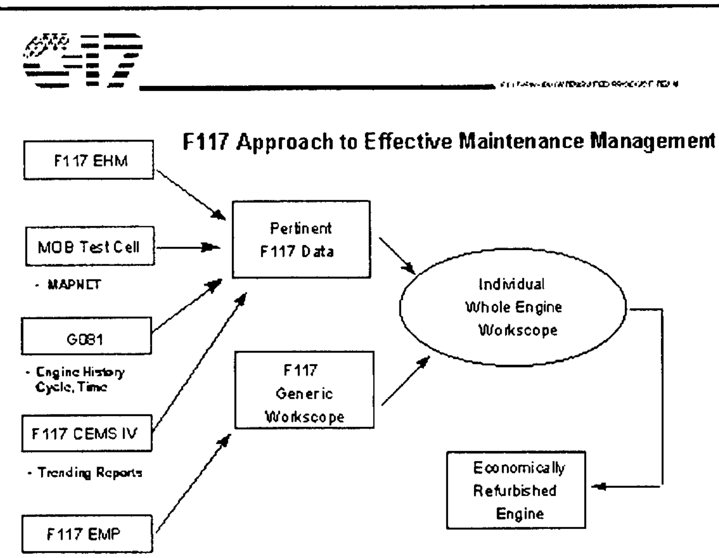



Figure 5 –Engine management plan

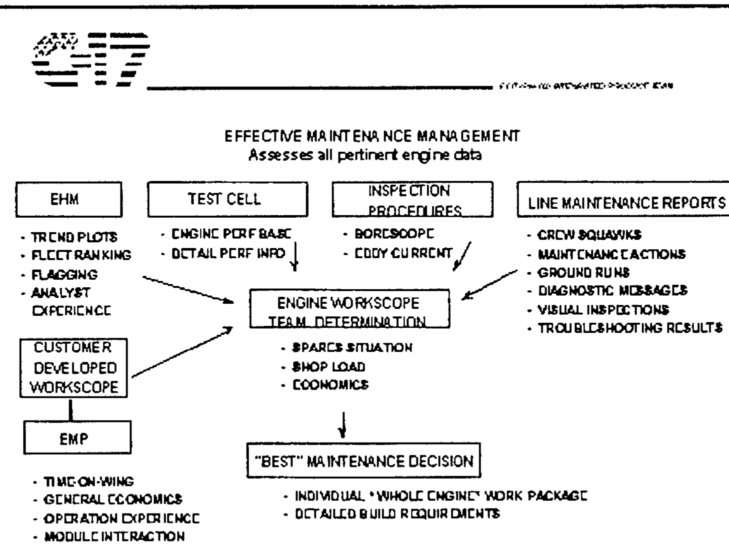



Figure 6 – F117 Individual work packaging

mathematically determines the risk over the life of a system or engine.

In general, the RCM analysis process begins by determining if a component is structurally or functionally significant. A separate RCM decision tree exists for each classification of part. This analysis is normally accomplished as part of the Failure Modes and Effects Criticality Analysis. The second step is to determine the failure consequences. The consequences are classified as:

- Safety;
- Economic/operational;
- Non-safety hidden failure;
- Safety-hidden.

After the failure consequence is determined, actions to avoid subsequent failures are proposed, and evaluated for applicability. The most appropriate is chosen. Cost-benefit analysis may be conducted to determine the optimum mix of new parts (old design), new parts (new design) or repaired/reworked parts. The maintenance options are:

- Do nothing;
- Redesign;
- Perform preventative maintenance tasks.

The preventative maintenance tasks include servicing/lubrication, on-condition inspections, hard-time rework or discard, and a combination of the previously mentioned options.

After applying RCM, the US Navy classifies turbine engine life limited parts as structurally significant items, which are managed through the preventive maintenance task of hard time discard actions, based on a safe-life limit.

A more complete explanation of the RCM analysis process is documented in ATA 143 (MSGIII - Recommended Maintenance Practices) and MIL-STD-2173. This document should be thoroughly reviewed before implementing the RCM process in any engine program.

## 3.4. LIFE MANAGEMENT PLAN

Each engine type, model, or series can benefit from a life management plan that lays the basis for the process appropriate to those engines. Typical life-management plans provide a structured, logical progression of activities that integrate the design, development, manufacture, use and repair or upgrade of an engine throughout its life. Such a plan will typically begin shortly after the beginning of development and include:

- Design Maintenance Concept
- Field Maintenance Capability
- Depot Maintenance Capability
- RCM Analysis Results

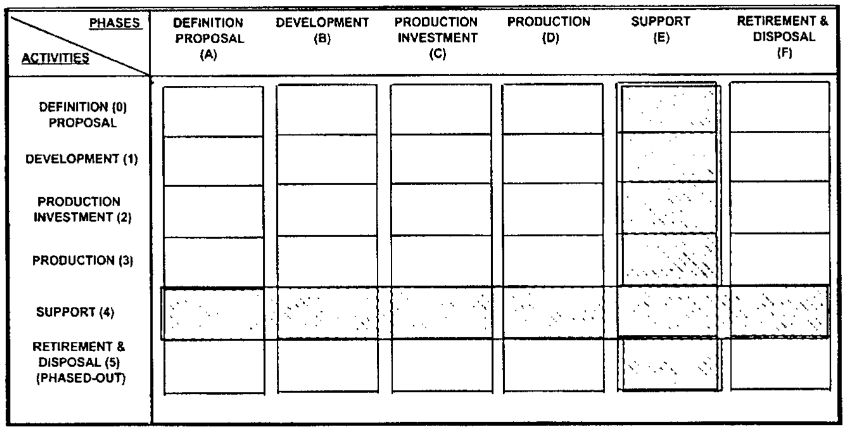



Figure 7 – Recurring cost of ownership

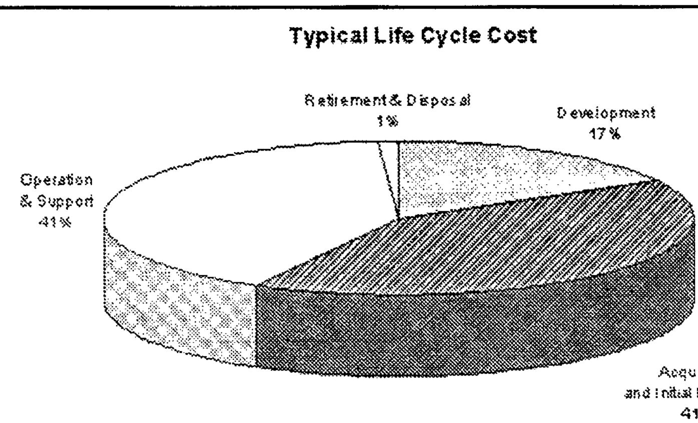



Figure 8 – Typical life cycle costs

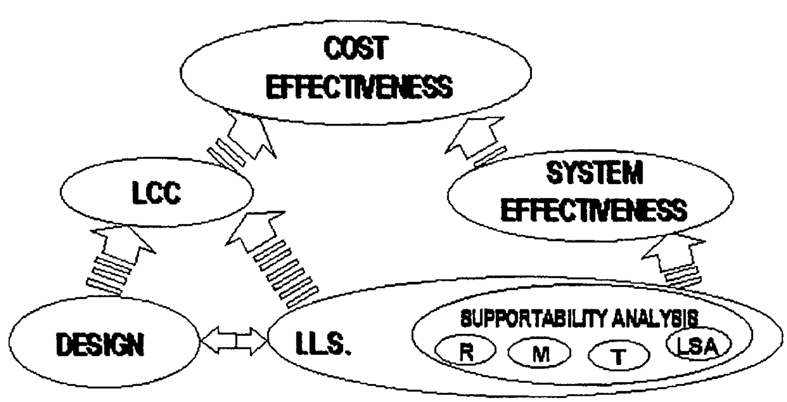



Figure 9 – Cost effectiveness dependencies

- Initial Life Estimates
- Age Exploration Plans
- Spares Requirement Process

### 3.5. MAINTENANCE PLANNING WORKING GROUP.

The life management plan will need to be updated periodically, as new data becomes available from either advanced analysis tools or field and test experience. Any changes in the maintenance concepts will also require the plan to be updated.

## 4. PHYSICAL BASIS FOR INSPECTION AND REUSE

There are two competing strategies for life management, the safe-life approach and the damage tolerance approach. Within the damage tolerance approach, there are two methods: standard ENSIP, and 'retirement for cause'.

The life defined in hours using the safe-life approach is often called 'hard time' or 'hard life'. It is based on a calculated or demonstrated life for a nominal component, given an estimate of the life history and material properties. Conservative factors are applied to the material properties or the demonstrated test life to allow a safety margin for the minimum-material-property component and extremes of operation.

The damage tolerance approach is dependent on the component being designed to have a relatively long crack growth (residual life) period, between a detectable flaw and rupture. This period determines the basis for the interval at which inspections must be performed, to ensure that the component is still serviceable, and to retire components that have flaws. The standard ENSIP method sets inspection intervals, but the ultimate retirement life is still a "hard time" life as detailed above.

The "retirement for cause" method includes damage tolerance analysis to set inspection intervals. However, retirement life is based on repetitive inspections until a flaw is found, or the part is retired for economic reasons. At some point in the life of a component, economics will dictate replacement of a component based on its "hard time". The mature engines addressed in this report, that were designed and developed prior to 1980, are managed to the safe-life approach except for the F100 engine that uses damage tolerance.

Two types of flaw may exist. These are surface cracks and buried flaws. Surface cracks are more easily detected by NDI methods. They may have a different inspection methodology and periodicity from buried cracks. Manufacturing processes and defect reduction techniques such as hot-isostatic pressing are the primary controls to reduce the frequency and severity of internal porosity (precursor to cracking) in blades and disks.

### 4.1. SAFE-LIFE

Engine designers' work hard to establish life limits for components that may threaten operational safety, should they deteriorate. The lifing procedure employed to establish life limits varies with component type, and the form of damage. In modern gas turbines, discs and spacers are normally designed to withstand LCF, burst by overspeed, and creep. This applies to both the hot and so-called cold sections of the engine as engine pressure ratios and compressor exit and combustor exit temperatures rise with more advanced designs. Turbine blades and vanes are designed to withstand creep as well as thermomechanical fatigue and high-cycle fatigue.

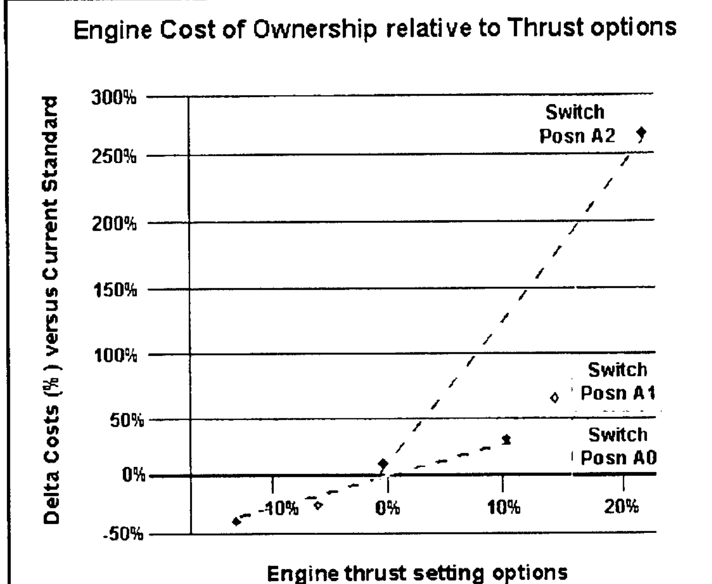



Figure 10 – The incremental cost of thrust

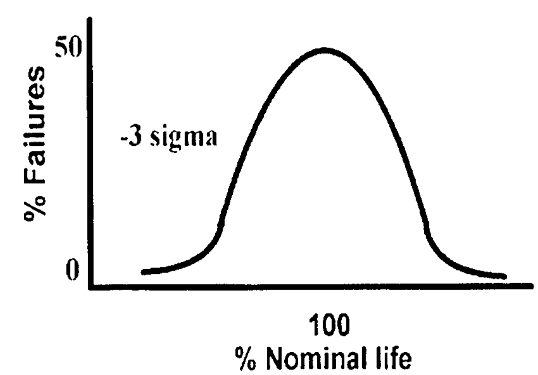



Figure 11 – Residual fleet life when first component is retired

The life limits for these components are initially based on estimates of in-service damage accumulation rates. The limits are later revised when field experience has been accumulated. For discs, the most common procedure for establishing safe life limits follows a 'time or cycles to crack-initiation' criterion. For turbine blades and vanes, life limits may be prescribed by engine designers. In particular, turbine blades and vanes in aero engines are seldom lifed because of the difficulties associated with predicting the service behaviour of metallurgically complex material systems under conditions that can vary widely with user practice. Furthermore, in the case of

coated components, these difficulties are made worse by coating substrate interactions that are not usually well understood. A 'life on-condition' approach is sometimes employed, for instance, where creep growth or untwist of airfoils is measured and distortion limits are used as retirement criteria.

As indicated, rotating parts such as compressor and turbine discs are usually life-limited due to LCF damage accumulation. The most common lifing method employed for these parts follows a 'cycles to crack-initiation' criterion. A minimum life capability is statistically defined for simulated service conditions through extensive testing. This normally includes standard coupon testing and component test verification in spin-pit test rigs. The statistical minimum is usually based on the probability that 1 in 1000 components will have developed a detectable crack (typically chosen as a crack 0.8 mm long).

The approach has been criticised as being overly conservative and costly on the ground that disks are usually discarded with significant amount of useful residual life. There are two major concerns when a cycle-to-crack-initiation rejection criterion is used to life rotating parts. The first one, by implication, is that 99.9% of the components will be retired before any detectable crack has formed. These parts may have a much longer service life before they develop a 0.8-mm crack. Secondly, the components may be capable of tolerating crack sizes much greater than 0.8-mm mentioned above. This limit reflects the sensitivity and reliability of current NDI methods rather than the mechanical tolerance of a particular part to cracks.

As can be seen from figure 11 a large percentage of components have useful life remaining (potentially many thousands of cycles) depending on the material and usage scatter) when the first component reaches its retirement life. Economical considerations would suggest that procedures which allow the use of components to attain their individual life rather than the life of the shortest lived component would greatly reduce the need for replacement components. Likewise total dependence on rejection by inspection for components with long manufacturing lead times and demand based purely on part rejection will lead to readiness issues at fleet lives approaching the nominal. The secret to cost efficient maintenance policies is to achieve a balance between parts life usage and readiness.

It is quite clear that the maximum life cannot be safely extracted from the whole fleet of components unless each component is considered, and has its life-usage tracked on an individual basis.

## 4.2. DAMAGE TOLERANCE (FRACTURE MECHANICS)

To ensure that the life potential of rotating hardware is more fully used, alternative lifing procedures have been suggested. These are based on damage tolerance and the application of fracture mechanics principles. The philosophy behind these alternative lifing procedures assumes that the component may be capable of continued safe operation during crack growth. This is dependent on the cracks growing sufficiently slowly during service to allow their growth to be reliably detected, and if necessary, monitored through regularly scheduled inspections. In practice, crack growth has only been 'monitored' when it has been necessary to overcome a shortfall in the required service life. It also appears that the 'monitoring' is usually restricted to a life extension for components that do not have a detectable crack.

In their most elementary forms, these alternative damage tolerance based lifing procedures, known as Life-On-Condition, Retirement-for-Cause or simply Fracture Mechanics lifing, assume the following.

- That the fracture critical locations of a component contain crack nucleation sites of a size that lie just below the detection limit of the NDI technique used to inspect the component;
- At some point in the life of the component it is assumed that a crack may have started, and regular inspections are begun;
- The crack then initiates, and grows during service in a manner that can be predicted by linear elastic fracture mechanics, or other acceptable methods;
- All cracks above a critical size will be detected, and monitored;
- When the cracks reach a predetermined size the component will be retired. In some cases, this will be the detection size.

It is clear that the successful application of damage tolerance based procedures depends on the supporting technologies. These technologies include non-destructive inspection, mechanical testing of test coupons and components, structural analysis, mission profile analysis and condition monitoring of components. Well-directed and extensive materials testing at room and elevated temperatures must be performed to obtain crack growth-rate data, for the application of deterministic and probabilistic fracture-mechanics based life-prediction concepts.

The basic hypothesis that allows a retirement-for-cause approach is that each component, as manufactured, contains inherent metallurgical features provide crack initiation sites, which must be allowed for in the design process. This assumption does not imply that the parts are defective, rather that natural crystallographic features occur when a component is cast or forged. Acceptance and understanding of this fact has been hindered by the inferences that components from a vendor or engine manufacturer are "defective" or "flawed". In fact, all materials have microporosity, lattice vacancies, grain boundary inclusions, etc. It is the size and frequency of these characteristics that separate a "good" component from a "defective" component.

The Engine Structural Integrity Program (ENSIP) for the USAF takes account of these factors, because it:

- Recognises the engineering nature of the materials;
- Characterises the inherent defect distributions;
- Provides for their inclusion into the design process;
- Establishes the level of inspection at manufacture to

assure the control of the processes that generate these defects;

- Establishes the residual life for each component.

### 4.2.1. NON-DESTRUCTIVE INSPECTION

The primary constituent of a life management policy based on inspection and reuse of serviceable components is to use an appropriate tool from the vast array of non-destructive inspection processes available today. While no single method is universally applicable, eddy current and fluorescent penetrant methods have found wide acceptance. X-ray, magnetic particle and neutron radiography, etc have practical but limited application. As can be seen in figure 12, once the initial flaw size has been determined for the *as-manufactured* condition an interval of failure free operation can be determined. This may be for either an actual initial material defect distribution (probabilistic) or assumed flaw distribution (deterministic). For many materials and processes, defect distributions have been developed from specimens cut up and analysed with microscopes. Assumed initial flaws are typically set at the stage I facet size based on historical observations.

Based on the crack growth characteristics of the material, environment, and loading, a crack growth forecast is then calculated. An interval is established to allow the non-destructive inspection (NDI) operation to just miss a flaw and provide a safe growth interval (one-half the crack growth life). Subsequent inspections determine whether such a flaw existed. If it did not, the part is returned to service for another interval. If the inspection detects a flaw, the part is retired.

Any inspection process inherently has a finite probability of finding a crack and therefore a finite chance of missing a crack. It is essential to be able to quantify the probability of detection (POD) to establish a reasonably effective inspection process for specific component critical areas, such as bolt holes, broached features, webs and other areas. The emphasis of this approach is on the largest crack that could be missed rather than the smallest crack that can be found. Inspection methods should be chosen with this in mind. Eddy current is a primary inspection technique for areas that rely on small flaw detection, and for those areas which have high residual compressive stresses. Other methods of inspection that are suitable for service use include ultrasonic and fluorescent penetrant inspection. The Reliability Centred Maintenance (RCM) process addresses the relationship between the probability of failure, its progression rate and the severity of its consequences, the probability of crack detection and the inspection interval required.

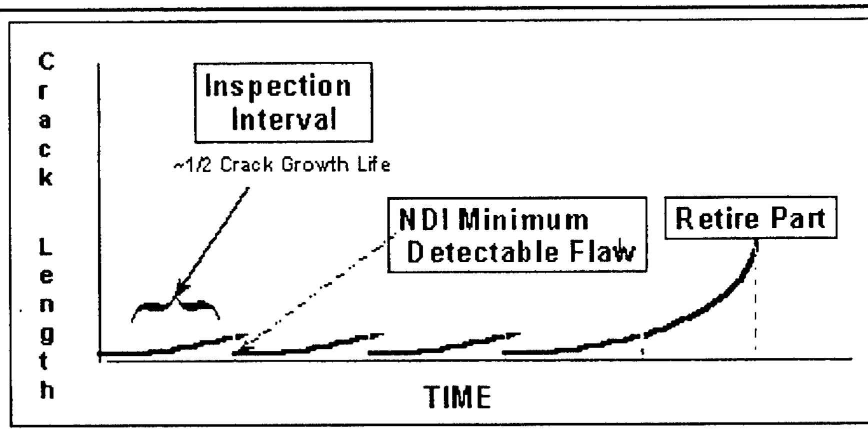



Figure 12 – Defect growth monitoring

If 'retirement for cause' is used then additional and more sophisticated ultrasonic and eddy current inspections will be necessary to ensure safety. Finally, it is important to combine the concepts of "hard time" with "retirement for cause" to minimise readiness concerns, and maximise safety. This means that all components, regardless of their crack-life history should have an over-riding 'hard life' at which they will be retired. In most cases, it is preferable to retire a fraction of the population before its actual life is consumed to preclude any in-service critical component failures and to provide production lead-time for replacement components. A rough guide would be to set a hard time at the point where a Weibull analysis shows about 10% of the fleet would be rejected.

### 4.2.2. INSPECTION REQUIREMENTS FOR DAMAGE TOLERANCE

MIL-STD-1783 or ENSIP requires non-destructive evaluations (NDE) of critical components during

- Manufacturing;
- Overhaul;
- Field inspection.

Designing with inspection in mind, and relating and limiting the (design) period of un-inspected service of all safety critical parts to the demonstrated inspection capability are parts of the ENSIP process. All safety and mission critical parts receive special inspection, compared to only rotating parts and cases under MIL-STD-5007E and rotating disks, shafts and spacers under FAR33.

Ultrasonic, fluorescent penetrant inspections (FPI), and eddy current inspection methods are relied on and qualified for the three different maintenance levels. NDE reliability demonstration programs to identify confidence limits, based on testing inspectors against known standards, are required under ENSIP. Verification of the inspection methods capability at each inspection site produces a probability of detection (POD) curve.

US experience is that FPI is capable of finding a 0.035 inch deep by 0.070 inch long flaw with good reliability for whole field inspections and a 0.020 inch deep by 0.040 inch long crack under focused inspection procedures. This is normally quantified in such terms as a 90% probability of detection, with a 95% confidence

level.

In a response to field related durability problems it is possible for the safe life practitioners to use enhanced inspections, such as eddy current, however this is not the standard practice for inspecting components. The verification and use of enhanced eddy current inspection procedures for flaw sizes (0.005 inch deep by 0.010 inch length) is a major difference between using the damage tolerance process and using the safe-life management procedures.

# 5. USAGE MONITORING IN LIFE DETERMINATION

There are several uses for life-usage monitoring equipment. The most basic is simply to prevent the failure of components in service, and to maximise use for a given cost. Secondary uses are to refine and adjust the design mission(s), mission profiles, mission mixes, operating conditions, etc. from those originally assumed by the designer. Better knowledge will always bring one of the following:

- Improved safety, through a reduction in usage;
- Reduced costs, through an increase in life;
- Security, in the knowledge that the life limits are correctly set.

## 5.1. FLEET USAGE MONITORING

It is a common experience for an operator to discover that a new weapons system with a new engine has much more capability than its predecessor and for new tactics, roles, missions, etc. to be developed for the aircraft. This is especially true for fighter/bomber aircraft and helicopters, and occasionally so for transport aircraft, leading to much shorter or longer lives than predicted by the designer. There are many examples that support this.

The F-16 is a lightweight multipurpose fighter/fighter attack aircraft and is in service in many NATO countries. The aircraft was developed with a version of the Pratt & Whitney F100 engine. In the original version, with the F100-PW-200, certain flight characteristics such as range, payload, operability, etc. were established. When the alternate fighter engines were developed, some characteristics were changed. With an F110-GE-100 engine, the number of practice bombing runs was increased due to reduced fuel consumption. However, the F110 was designed using the mission profiles, mixes, etc. from the Pratt & Whitney engine and some early problems arose. These were traced to the operational differences made possible by the better fuel consumption and subsequent changes to the training mission profiles.

A fleet survey was accomplished, somewhat belatedly, to identify these characteristics and create a new definition of the usage for accelerated mission testing (AMT). This usage definition was then applied to both the increased performance engines - F100-PW-229 and the F110-GE-129. It has proven to be a good indicator for the F16 aircraft, but not for the F100-PW-299 engine installed in the heavier, ground attack F-15E. In this case, the correct amount of time at high power settings was not included and cases of thermal distress that have arisen in fleet operations was not predicted.

Early and repeated (~3-5 year) monitoring of the fleet usage, with simultaneous recordings of significant engine parameters (rotor speeds, operating temperatures, etc.) and aircraft parameters (Mach number, altitude, attitude, etc.) must be done to identify operational changes. This is so that timely adjustment to life analysis and the subsequent maintenance and logistics provisions may be made.

## 5.2. PARTS LIFE TRACKING

There are many methods used throughout the NATO nations for tracking the life of fracture critical rotating components.

The simplest is to maintain a paper record of the hours of operation of the engine in which the component is installed. The remaining life of the components is recorded as the engine is assembled and then debited at periodic intervals as the engine is operated. When the stated life is reached, the engine is disassembled and the component replaced and retired.

A more complex, but still simple method accumulates cycles by a conversion of operating time and mission to cycles by a fixed algorithm. The conversion factors are sometimes known as $\beta$ (beta) factors. The process is then the same as for operating time.

A more complex method is for an on-board processor to measure cycles and time in certain temperature bands. This is then downloaded to a ground based computer system that maintains the configuration records and debits the life remaining on the tracked parts. Operational difficulties may occur with extended deployments away from the ground system and lead to a loss of cycles.

A further enhancement is for the data to be recorded in bulk, and then to be downloaded for analysis on a ground based system. This allows the full manufacturers design model to be used for analysis if required. This provides the most technically accurate approach.

The final enhancement is for the records to be updated in near real time by the on-board processing system as well as maintaining the configuration records. This is periodically downloaded to ground systems for planning use and avoids the deployment data loss. Usually a reduced version of the manufacturers design model will be used to extract the major and minor cycles and sum them for each monitored stress feature.

While the objective of these systems is to protect flight safety through the retirement of life limited parts prior to fleet failure, it is unclear that any one demonstrates a dramatic advantage in reducing the cost of operations. The operating risks, however, change significantly and chapter 9 provides a tool that will help operators to quantify these changes.

# 6. RETIREMENT STRATEGIES

Retirement strategies generally fall into two categories; predetermined intervals and retirement for cause.

### 6.1. RETIREMENT AT PREDETERMINED INTERVALS

This strategy is a conventional fatigue design approach for part retirement. It is based on low cycle fatigue (LCF) life. The LCF limit is established based on a lower bound(1 in 750 or 1 in 1000) distribution of crack initiation times. The use of the lower bound minimises the occurrence of cracking and the need for component repair. Implementation of this conventional strategy requires 100% hardware replacement at the lower bound LCF limit. Damage tolerance control philosophies could also be combined with this strategy as a means to ensure hardware structural integrity during the LCF life.

### 6.2. RETIREMENT FOR CAUSE

With the previous approach, only 1 component in a population of 1000 components would typically have initiated a crack, and the remaining 999 components would be retired with substantial useful life. By using the retirement for cause (RFC) methodologies, this useful life can be made available. This because each component is used until a crack is found in it – or a predetermined percentage of components have been rejected and one is effectively climbing the right side of the reliability bathtub curve. Substantial development of fracture mechanics concepts over the last several years makes it possible to predict crack propagation rates accurately enough to use these techniques.

RFC methodologies apply the fracture control based philosophy described in section [7.3]: All components would be inspected first at the end of a safety limit period divided by an appropriate safety margin (SL/SM) which would maintain the equivalent risk used in the conventional design life. It is important to note that in order to maintain an equivalent risk, there will be an increase in the number of areas to be inspected over the design life safety limit inspections. This is a natural outcome of the part reaching its predicted LCF life, and increased risk would occur if additional areas were not inspected. Only those components containing defects equal to or greater than $a_i$ (initial crack length) would be retired. All other components would be returned to service with the assumption that, if they have a flaw, it is less than $a_i$ and thus good for another inspection interval. This ensures that the crack propagation life is continually reset to a safe value, and that components are rejected only for cause (cracks or defects). Each component is allowed to operate up to its own specific crack initiation life (or a limit based on the fleet history of space cracking). If a crack is missed at the inspection interval, another chance will exist to find the crack when it is larger, without the component failing prior to that inspection. Clearly not all fatigue-limited hardware can be handled in this fashion:

- Components must be designed to have a slow crack propagation rate;
- The inspection interval must be long enough not to put undue constraints on weapon system readiness;
- The cost of the overhaul, and inspection of the hardware, must be economical and not negate the advantage of the life extension.

### 6.3. INSPECTION

Developing inspection processes for RFC components is a major undertaking. The inspection techniques must be able to find cracks emanating from internal inclusions and be detectable from any orientation in a finished part. In order to achieve this, new inspection standards, from which to measure the adequacy of new inspection techniques, must be designed and fabricated with buried flaws. The USAF has an inspection system and a set of standards, which were developed for this purpose.

An alternative is to use RFC for surface flaws only and to retire hardware at the buried-flaw inspection life-limit. This approach will still yield a significant extension in life over the historic minimum LCF life criteria. To maximise the gains it will be necessary to ensure that the buried flaw life of the weakest component in the fleet is controlled by design.

It is clear that in applying RFC, the non-destructive inspection capabilities are critical in defining the economic and physical inspection feasibility. As shown earlier, the crack length determines the residual life. Hence, the inspection interval and crack detection is limited by the capability and reliability of the inspection system. Inspection capability and economics are equally important considerations when implementing RFC on a particular component.

## 7. DAMAGE TOLERANCE FOR RISK MANAGEMENT

Damage tolerance analysis may be used, for components operating under safe life methods, for two reasons:

- Life extension or the usage of components beyond calculated low cycle fatigue limits;
- To manage a life shortfall

The use of damage tolerance to extend life limits was discussed in the previous section. With a life shortfall, crack growth lives are calculated from the capability of the inspection method (eddy current, ultrasonic, or FPI). A risk management plan is developed to keep risk to an acceptable level based on:

- How many cycles the fleet has operated;
- How many failures are likely to occur and have occurred;
- When the component in question can be inspected;
- How long it can be safely operated between inspections and before a repair or component replacement is required.

Because of ENSIP requirements, the application of these damage tolerance methods is part of the process and therefore easier than for a safe life design. Under ENSIP, provisions are made for addressing such problems through damage tolerance analysis, focused inspection, ease of engine disassembly, maintenance implementation plans, and risk assessment.

## 8. Overview of Current Maintenance Strategies

Appendix 1 provides information that is current at the time of writing, and which does not reflect advances that will surely occur for the newest engines including the F119-PW-100, F119 derivatives, F120-GE-100 and the EJ200.

## 9. Recommendations

- A systematic approach to component life management and maintenance is essential if the cost of ownership is to be reduced.
- Lessons can be learned from other operators. Conclusions should not be reached without investigation.
- Tailor the management of component life to the maintenance concept established by the end user in consultation with the engine manufacturer and the services logistics system.
- Engines that have a digital control system or digital condition monitoring system can benefit from an on-board life calculation algorithm presuming the range of operation is within the correlation range of the algorithm. Off-board processing may allow for a more complex algorithm.
- Usage monitoring equipment is particularly useful in determining differences in life consumption between operational bases, missions and aircraft series. Monitoring is more accurate than mission logs, sortie pattern factors, etc. that generalise the mission profiles. Experience has shown that differences in usage – which may have the same cyclic content – can precipitate failure of bearings, blades, etc. that would have been avoided with usage monitoring and corrective action when differences were detected.
- TACs collect more coarsely granulated data than usage monitoring systems, which catch and account for all turning points. However, when TACs are combined with the ENSIP approach the in-service safety levels of the two approaches – damage tolerance and safe life – appear to be virtually identical.

## 10. References

*Reliability-Centered Maintenance*. F Stanley Nowlan and Howard F Heap of United Air Lines. The reference number is MDA903-75-C-0349, dated 1978

*Maintenance Management - A New Paradigm* was published in leading maintenance journals in the United Kingdom, the United States and Australia in 1996. It is based on a paper of the same name presented at the annual conference of The American Society of Maintenance and Reliability Professionals in October 1995.

*Reliability-centred Maintenance* by John Mowbray, Butterworth-Heinemann Ltd, 1991, ISBN 0 7506 0230 9

# Chapter 4
## Modes of Gas Turbine Component Life Consumption

by

*(C. Eady)*

Page

## 1. INTRODUCTION

In this chapter the main failure modes of mechanical components are shown in pictures of real components, and described in simple physical terms.

Critical components, for which safe operating lives are defined, include the major rotating parts, the turbo-machinery disks and shafts, and structural casings subjected to high loads. In addition to the major components, other components in the rotating assemblies, such as spacers, cover-plates and seals, may also be designated as critical. To avoid unacceptable risk of catastrophic failure it is necessary to monitor the life usage of critical components and retire them from service before their allocated life has been exceeded.

A thorough understanding of the failure mechanisms affecting gas turbine components is essential if the failure modes, the safe-life, and the life usage of each component are to be accurately determined and safely monitored. Those failure mechanisms are:

- Low cycle fatigue;
- High cycle fatigue;
- Thermo-mechanical fatigue;
- Creep;
- Overstress;
- Corrosion;
- Erosion;
- Fretting and Wear.

The ability of a component to resist the effects of any of these failure mechanisms is a function of the material properties, the component design and the operating environment. These features of the component are effectively fixed by the design and application of the engine and cannot be affected by the way in which the engine is operated or maintained. As such the chosen engine cycle, its configuration and the selected operating environment may be considered to be intrinsic factors which impact on the rate of component life usage. Conversely, there are external factors which also have an influence on the rate of component life consumption but which can be reduced during engine manufacture, operation and maintenance. Best maintenance practice, no matter how good, can at best only restore component and system performance to the levels that were intrinsic to the original design.

The external factors affecting life usage rate are:

- Manufacturing and material defects;
- Build and maintenance errors;
- Foreign Object Damage (FOD);
- Limit exceedances.

Since these four engine damage sources are effectively under the control of engine manufacturers, operators and maintainers, they could be described as avoidable. In reality, however, this is difficult to achieve.

The accuracy of the process for determining safe component life and the monitoring of its consumption may have a major impact upon the safety of the engine or on its life cycle cost. Engine design-features and operational environments that most undermine safety or are most costly, financially or in maintenance effort, must be identified. Only then can managers assess the potential benefit of proposed modifications to the design or changes to operating practices. Even minor changes to operating practice, which may have little or no operational impact, can significantly lower life usage rates and bring the benefits of higher availability and reduced life cycle cost.

Monitoring systems can only monitor phenomena that they can detect and measure. Therefore, great care is required during the design and introduction of any system, to ensure that it meets the needs of people at all levels in the operator's organisation.

## 2. GAS TURBINE ENGINE FAILURE MECHANISMS

### 2.1. MECHANICAL FATIGUE

In most practical aerospace engineering situations, the applied stresses are not steady. Instead, the loading fluctuates, often in a random manner. Under conditions of cyclic stress, it is frequently found that failure occurs at lower stress levels than would be expected, were a steady stress applied. This phenomenon is fatigue and causes the vast majority of in-service failures.

The process of fatigue is usually divided into the following 3 phases:

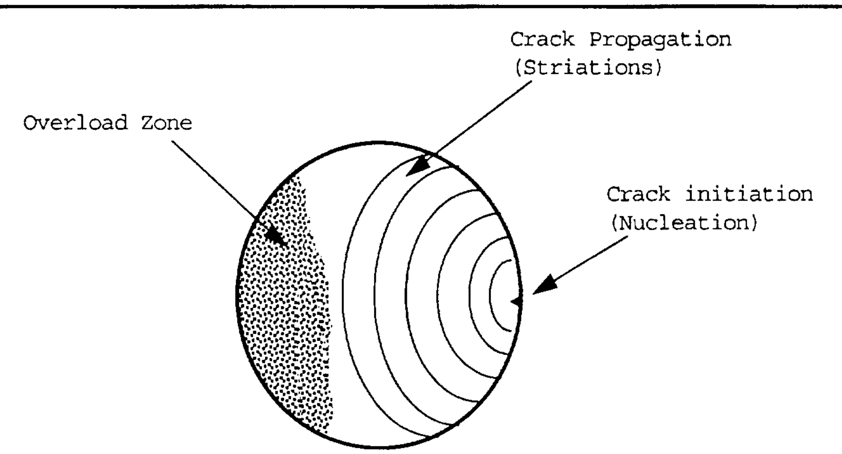



Figure 1 - Characteristic fatigue fracture

- Crack initiation (primary stage);
- Crack propagation (secondary stage);
- Unstable crack propagation (tertiary stage).

Figure 1 shows the features of a characteristic fatigue fracture. The number of stress cycles to failure is dependent upon the material type, component processing, load history, design, and the degree of loading.

The fatigue behaviour of a material is normally plotted on

an S-N graph. Here S is the average stress range applied (in simple harmonic motion), and N the number of cycles to failure, at that combination of mean stress and stress range. Fatigue failure is a statistical phenomenon and accurate measurement of the fatigue properties of a material requires a large number of tests to ascertain the failure probability distribution at each applied mean stress and fluctuating stress amplitude. A characteristic S-N plot for zero mean stress, showing the mean failure curve and the failure distributions, is given in figure 2.

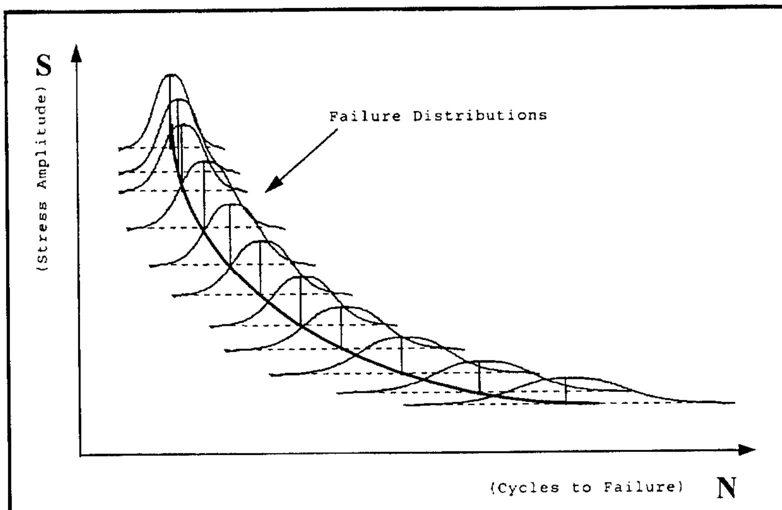



Figure 2 - S-N diagram.

Component fatigue life is not only dependent upon the choice of material and the magnitude of the mean and cyclic stresses. Materials production and processing can have a significant impact on component performance in service. Microstructural variables such as changes in grain size, alloying and the presence of non-metallic inclusions will affect the fatigue life. Fatigue behaviour is also very sensitive to:

- Random stress fluctuations;
- Stress concentrations;
- Surface finish;
- Residual stresses;
- Corrosive environments.

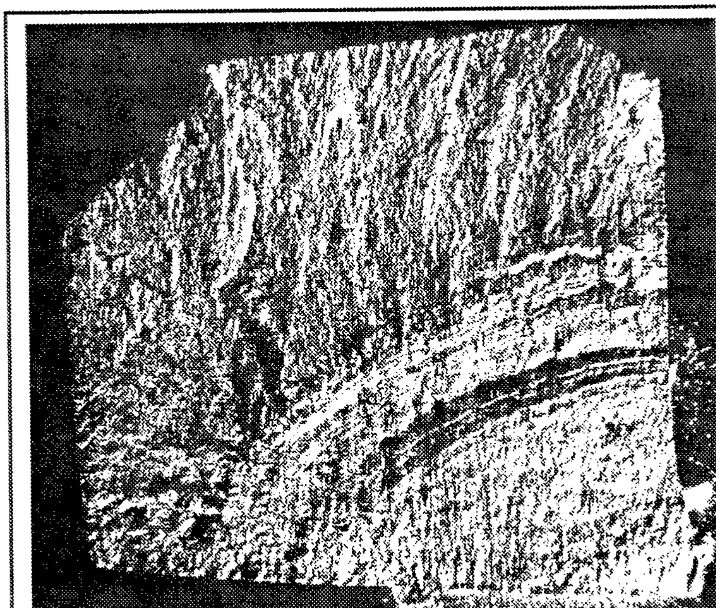


Figure 3 - Fatigue crack growth

The effect of the above factors is that severe fatigue life penalties can be imposed through inadequate component design or inappropriate material selection for a given operating environment.

The design loads for gas turbine components are mainly in the elastic region of the stress/strain curve of the material. In some cases, cyclic softening can cause highly localised plastic deformation. This happens when high stress concentrations cause yielding, which then reduces the local stress to levels where yielding no longer occurs.

The traditional approach to fatigue, for which the S-N curve was designed, chiefly concerned itself with failure after a large number of cycles. At higher stresses and lower cycles, the fatigue life reduces progressively and the scale of the plastic deformation creates difficulty in interpreting test results. In this region of Low Cycle Fatigue (LCF), sometimes termed high-strain fatigue, the materials tests are based on constant strain cycling rather than constant stress cycling.

The transition point from LCF to High Cycle Fatigue (HCF) is generally assumed to occur where the total strain comprises equal proportions of elastic and plastic strain. However, for convenience, LCF is often considered to lead to failure in less than $10^5$ cycles and HCF to lead to failure after more than $10^7$ cycles. The intermediate range may be considered to fall into either region, depending on the design application. It must be remembered that it is not the number of times that a cyclic load is applied that is of particular importance. Rather the amount of damage accumulated during each application of the load. For critical gas-turbine components, it is appropriate to differentiate between LCF and HCF by identifying the driver of the cyclic loading. LCF is typically driven by pilot demands and the application of relatively large loads whereas HCF is typically driven by sources of vibration and the application of relatively small loads.

### 2.1.1. Low Cycle Fatigue

If the amplitude of the cyclic stresses applied to a component is very high (LCF), the accumulated strain energy per reversal will be significantly higher than for an HCF loading cycle. For this reason, under LCF, a component will spend a very small proportion of its life in the primary, crack initiation stage and the majority of its life in the secondary, crack propagation phase of fatigue failure. As a result, since crack initiation is often governed by random imperfections in the microstructure, the scatter in LCF data is lower than that for HCF data, as can be seen in figure 2. Under such high loading, it is also

possible that plastic deformation around a concentration feature will redistribute ('shake down') the stress to a less damaging amplitude and leave the component under residual compressive stress whilst stationary.

Most of the critical components within gas-turbine engines, the turbo-machinery disks and shafts, and pressure vessels such as the combustion chamber outer casing, are subjected to very high loading cycles and are, therefore, life limited by LCF. The mechanical loading on these components is caused by:

- Centrifugal forces and thermal loads on the disks;
- Torsion and bending forces on the shafts;
- High pressures within the casings;
- Thermal gradients within components

The cyclic nature of these forces is due to variations in engine power setting. One complete, or major, cycle is experienced when the engine is accelerated from standstill to maximum engine rotational speed and then returned to standstill. Minor cycles of varying size are experienced for all other throttle movements. LCF for a gas turbine can therefore be characterised as loading cycles caused by variations in rotational speed, temperature distribution in parts, or engine internal pressure, which are most often due to throttle movement. The appearance of typical LCF failures is shown in figures 3, 4 and 5.

## 2.2. HIGH CYCLE FATIGUE (HCF)

Fatigue failure occurs when either the material fracture toughness is exceeded by the combination of applied stress and crack size, or a critical crack size is attained in a highly stressed region of a component. The process to achieve either of these conditions involves crack initiation and sub-critical crack propagation. Whilst components under an LCF regime spend the majority of their lives in the crack propagation phase, for HCF, crack initiation often tends to be the time-consuming process. Low amplitude, high frequency loading cycles (HCF) could quickly propagate a LCF initiated crack to failure.

HCF is primarily a function of engine design. A component that fails in HCF does so because it has been subjected to a large number ($>10^7$) of stress cycles (where compressor blades are subjected to high cycle aerodynamic loading this is known as flutter). A number of factors can cause high frequency loading of components. These are generally known as drivers and their causes are listed below.

### 2.2.1. ENGINE CONFIGURATION

The layout of the engine itself is the cause of many HCF drivers. The presence of both upstream and downstream obstructions in the gas path can create perturbations in the flow that will cause the turbo-machinery blades to deflect each time they pass. The passing frequency depends upon the number of obstructions and the rotational speed of the stage. For example, each blade on a turbine assembly rotating at 19400 RPM behind a set of 34 nozzle guide vanes will experience 39.6 million loading cycles per hour. It is, therefore, important that the stress amplitude is very small so that the blades are operating below the fatigue limit set by the curve in figure 2.

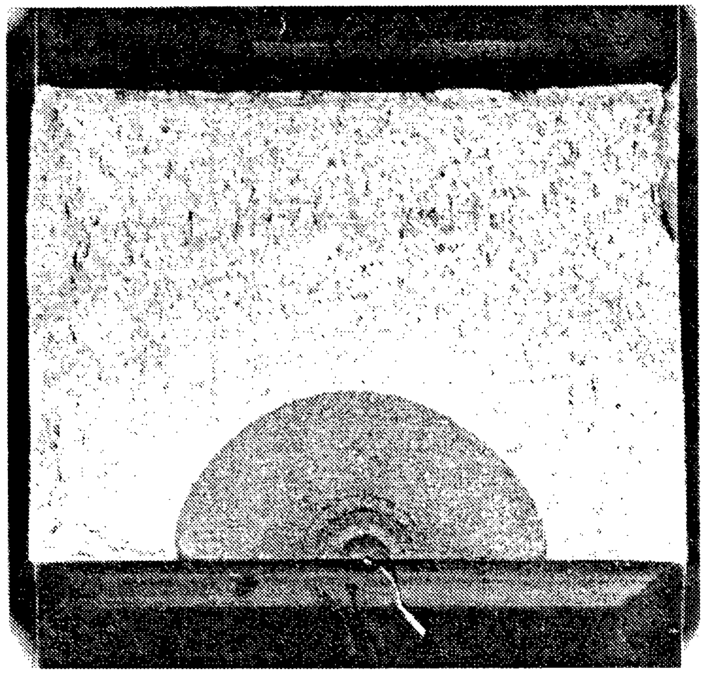


Figure 4 - Fatigue crack growth

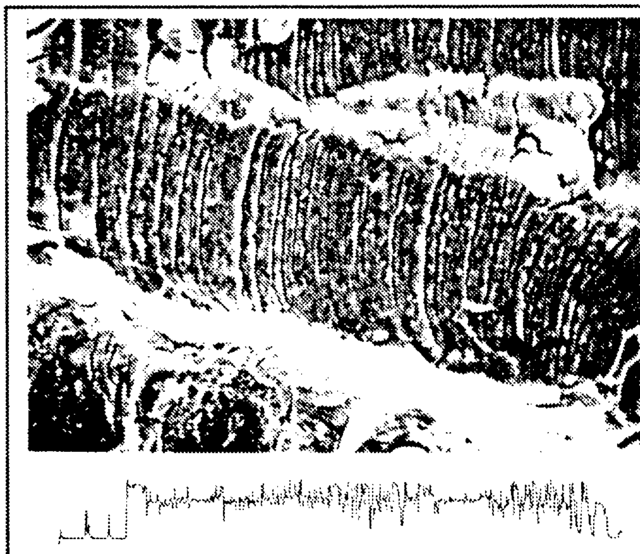


Figure 5 - Fatigue failure in an LP blade attachment lug

Campbell Spoke Diagrams are used to identify regions where problems might occur. A blade design has its natural frequency calculated or measured. Corrections are applied to account for a natural frequency drop due to temperature rise and the stiffening effect due to radial centrifugal forces. The overall effect is usually for the natural frequency of the blade to rise as engine speed increases.

An illustrative Campbell Spoke Diagram is at figure 6. It shows the frequency of vibration of various components against engine speed. Included in the diagram are the first seven Engine Orders (EO) and the first flap (1F) and second flap (2F) vibration modes of a typical blade. Due to the higher energy involved, engine speeds above 75% (5355 RPM) and the first 3 engine orders give the greatest cause for concern. In this example, the first flap and the second engine order interact between the 100% (7140 RPM) and the 110% overspeed (7854 RPM) conditions. This is therefore not a viable blade and re-design is necessary to alter the natural frequency of the blade.

The potential resonance caused by interaction between the second flap mode and the 6EO and 5EO spokes is unlikely to be of high energy. However, it may be significant if there is a strong upstream excitation. In which case, the resonance could be driven to high energy levels by splitters or support struts in sets of 5 or 6 or sub-multiples thereof.

### 2.2.2. INTAKE EFFECTS

One of the strongest HCF drivers can be caused by intake distortion of the airflow which creates a high energy, normally first engine order (low-pressure shaft speed) driver. Intake distortion can be inherent in the design or due to the operation of the aircraft. Bifurcated intakes often create uneven pressure and velocity distributions at the compressor face that induce flutter in addition to second engine order excitation.

In operation, aerodynamic effects such as intake masking during high alpha (angle of attack) manoeuvres or during high yaw or side-slip manoeuvres can also grossly distort the inlet flow. Supersonic intakes, when operating at off-design conditions, also cause distortion at the front face of the engine. Alternatively, distortion can be caused by physical blockage. Such might occur in icing conditions, or if a large object is ingested. Fortunately these events are normally of short duration and hence rarely lead to engine failure. However designers must allow for some element of running with distorted intake airflows and operators must try to avoid these situations wherever possible.

### 2.2.3. OUT OF BALANCE

It is impossible to perfectly balance the rotating components in a gas turbine engine. A mechanical system that has heavy components rotating at high speed will experience vibration. An out-of-balance rotor will create a vibration driver proportional to the shaft speed and will excite components across a wide range of frequencies in normal operating conditions. Components must therefore be designed to avoid the natural frequencies that are likely to be generated by an out-of-balance rotor.

## 2.3. THERMO-MECHANICAL FATIGUE

Thermomechanical fatigue damage is caused by a combination of the external loads and cyclic compressive and tensile loads induced by thermal gradients across components.

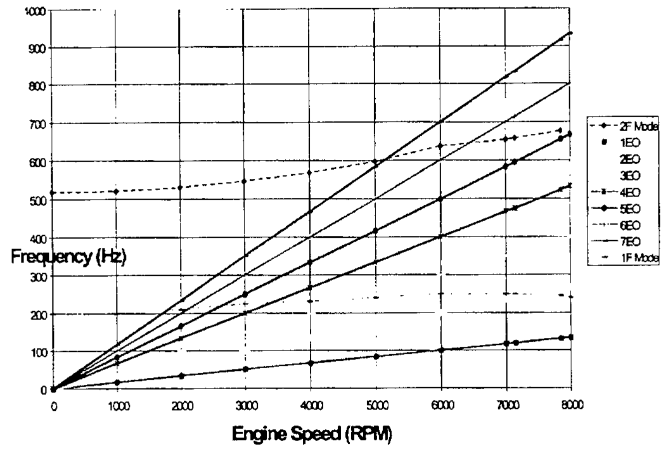



Figure 6 - Campbell 'spoke' diagram

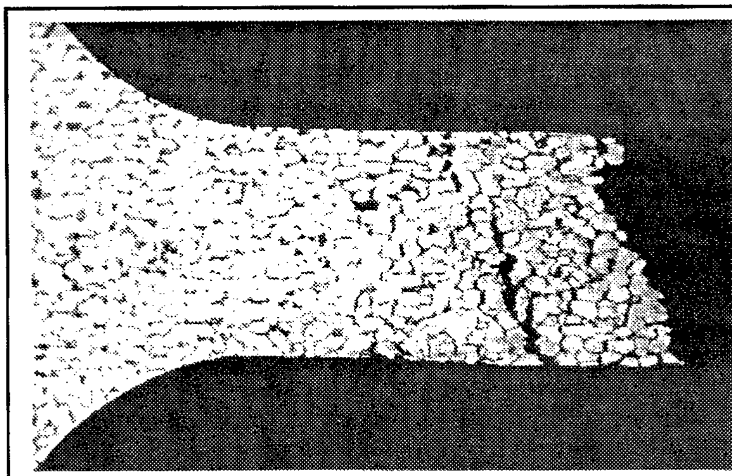


Figure 7 - Intergranular creep failure in nickel superalloy

This effect is particularly significant for turbine blades and vanes, and especially cooled blades and vanes. Taking an uncooled HP turbine blade as an example, before first engine start it has no residual stresses and is at uniform temperature. Upon engine start the blade experiences high temperatures on its outer surface whilst taking longer to reach operating temperature at its core. This leads to excessive compressive forces particularly at the leading edge and the material plastically deforms to relieve the stress.

As the blade core reaches steady state temperature, the outer surface goes into tension due to the earlier compressive deformation. The centrifugal loading on the blade increases this tensile loading. As the throttle is retarded, the centrifugal forces reduce and the outer surface cools faster than the core causing the outer surface to try to contract. This contraction is opposed by the core, which holds the outer surface in tension.

When the engine is shutdown and reaches the ambient temperature throughout, there remains a tensile stress in the outer surface of many components. This process enhances the level of tensile stress cycling experienced by the blade, particularly in the thinner sections such as the leading and trailing edges, where temperature changes occur most rapidly.

It can therefore be seen that the blade goes through a stress sequence with every change of temperature. The scale of the load imposed is proportional to the temperature gradient induced in the blade and this is a function of the rate of throttle movement. Therefore, for long engine life throttle movements should be made as slowly as possible.

Thrust producing engines, such as turbo-jets or turbo-fans, operate at varying speeds and temperatures. The thermal and mechanical load cycles tend to happen simultaneously. However, since the thermal loading affects the degree of mechanical loading, engine life usage counters really need Thermal Transient Algorithms (TTA) in order to most accurately monitor fatigue life consumption.

Some torque producing engines such as turbo-props and turbo-shafts, may have one shaft running at constant speed whilst the temperature varies. In such cases, the mechanical load cycles are easily monitored. On the other hand, changes in torque demand are achieved by altering engine-running temperature. Hence, thermal fatigue effects predominate and must be monitored, to ensure safe and efficient engine operation.

For critical components such as turbo-machinery disks, the transient thermal effects may only be significant during major throttle modulations. However, the disks do suffer a temperature gradient from bore to rim. This gradient is usually a cooler bore to a hotter rim, especially for turbines and the HP sections of compressors, but it may go the other way for LP compressors with anti-icing air or balancing air within the shaft. Whichever way the thermal gradient operates, it will also effect material properties. The thermal gradient will produce a stress field that varies across the radius due to the temperature differences and the coefficient of expansion.

## 2.4. CREEP

Creep is a time-dependent inelastic deformation of metals and alloys, which occurs under stress, at temperatures above about 0.5 of the melting point temperature. The higher the temperature and the greater the load, the faster will be the rate of deformation. In a gas turbine the components which suffer most from creep are the hot parts of turbine disks and the turbine blades. However, blades tend to fail due to impact with adjacent rows or casings rather than purely due to creep. Figure 7 shows a typical failure.

Figure 8 shows the three stages of creep.

- The Primary Stage depicts rapid extension at a decreasing rate; this is of interest to the designer as it forms part of the total extension reached in a given time and so affects the choice of clearances.
- The Secondary Stage shows creep occurring at a relatively constant rate; this is the important part of the curve for most applications.
- The Tertiary Stage shows an acceleration of the creep rate to failure; this stage should be avoided in operation but the transition from the Secondary Stage to the Tertiary can be difficult to predict.

### 2.4.1. STRESS RUPTURE

For design purposes, so-called stress rupture curves are used to simplify creep modelling. These are obtained by measuring the time to failure of a specimen under constant load and temperature conditions. The curves are simply a plot of the log of time to failure against temperature for various loads. The result is usually a straight line and this representation simplifies the calculations.

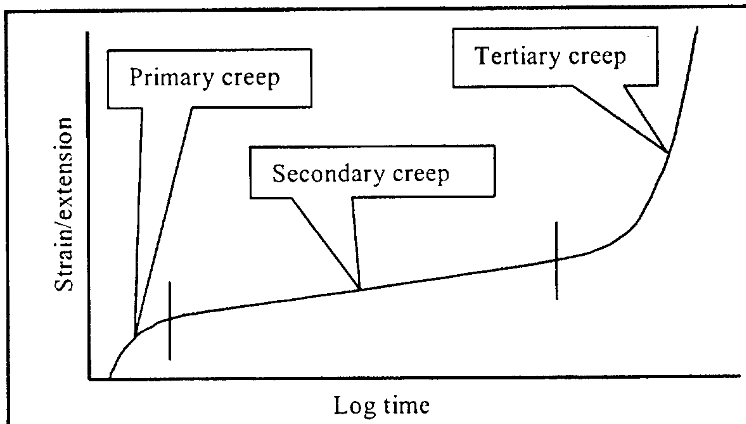



Figure 8 – The three stages of creep

## 2.5. CORROSION

Irreversible damage can be caused to engine components in aggressive environments. Corrosive elements can be introduced to the engine in the fuel (eg sulphur) and in the inlet airflow (eg airborne pollutants or sodium chloride from marine environments).

Corrosion can be controlled through careful selection of materials, the application of coatings, and the maintenance of the surface finish throughout the life of the engine. Regular water washing after each sortie, if necessary, should be considered for engines operated in corrosive atmospheres.

Corrosion may reduce the life of an aero engine in 3 ways.

- By reducing the load bearing area. Corrosion causes loss of material, and can weaken a component so much that it fails under normal loading. Corrosion on this scale is usually easy to detect during routine inspection. Much more difficult to detect is a small corrosion pit that acts as a stress raiser hence causing a premature LCF or HCF failure. Any breakdown in sealing effectiveness within the turbine section may also permit hot gas ingress to the disk void areas and result in material degradation.
- By reducing aerodynamic efficiency. If aerodynamic surfaces, particularly blades and vanes and to a lesser extent walls and diffusers, suffer corrosion

they are roughened. The effect of this is to alter the nature of the airflow over them and hence reduce their efficiency. Any reduction in efficiency of the turbo-machinery will result in a loss of thrust at a given cycle temperature or require that temperature to be raised to maintain thrust. In either event, the life of the engine will be reduced due to premature rejection for low thrust or excessive operating temperature.

- By causing a blockage. With aluminium based materials, the products of corrosion can occupy six times the volume of the non-corroded material. Therefore, in extreme cases, the corrosion will result in severe restrictions to cooling passages and possible bursting of thin-walled sections. The disruption of cooling flows may also cause hot spots to develop and lead to early creep failure.

Although corrosive action can remove enough material to directly affect the performance of a component, the mechanical integrity of high strength components is likely to be jeopardised long before any visible evidence of corrosive attack. This is due to the creation of sites from which fatigue cracks can propagate. Examples of the effects of corrosion are shown in figures 9, 10 and 11.

### 2.5.1. CORROSION FATIGUE

The response of different alloys to combined stress and corrosion varies greatly but those that fail usually do so without any gross corrosion. When there is a combination of a corrosive environment and a fluctuating stress, a hazardous condition described as corrosion fatigue can occur. The effect of corrosion fatigue is usually to reduce the fatigue life much more severely than would be expected by the separate consideration of the two mechanisms.

The first stage of corrosion fatigue is usually pitting of the surface layer of the material. This weakening allows micro cracks to form and the second stage establishes corrosion in these cracks. In this way, both the crack initiation phase and the propagation phase of fatigue failure are accelerated by the corrosive process as good material is removed and the cracks are wedged with the products of corrosion. Electrolysis at a crack tip may cause atomic hydrogen, which can penetrate the metal lattice and cause hydrogen embrittlement. This may further accelerate crack growth.

## 2.6. EROSION

Erosion is the cumulative damage to components in the airstream caused by small hard particles carried in the gas path. What differentiates erosion from any other damage mechanism is the scale and nature of each individual damage event. Individually each event is inconsequential in terms of performance loss or reduction in mechanical strength. However, when very many events take place at one particular area of a component then the performance or strength of that component can be severely compromised. Erosion is often most apparent in compressors, particularly those used in aircraft that routinely operate in dusty/sandy conditions. Helicopter engines are the worst affected, since by their very nature, they are prone to operate in dust clouds generated by the rotor downwash.

The typical effects of erosion within a gas-turbine engine are:

- To remove material from the rotor tips increasing tip clearance and reducing performance;
- To remove material from the leading edge of turbine blades as shown in figure 12;
- To reduce the chord width of the blades thus reducing aerodynamic performance.

Figure 9 - Salt corrosion of PT blade of aero-derivative gas turbine in marine environment

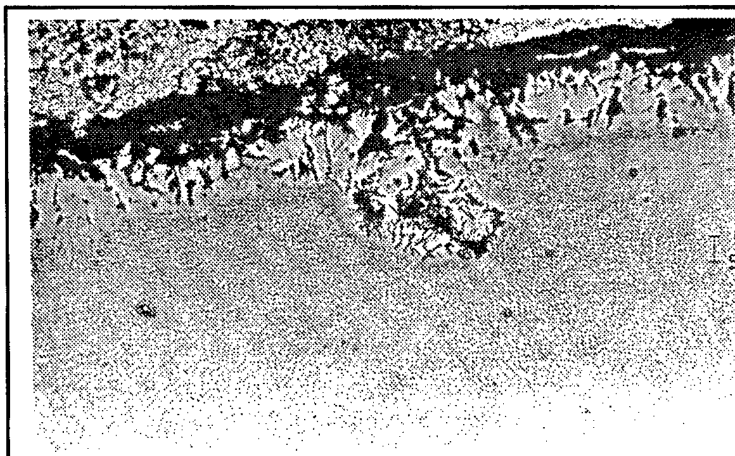


Figure 10 – Corrosion in nickel superalloy

- Due to the vulnerability of helicopter engines to erosion, many are fitted with inlet particle separators, which, at the cost of some engine performance, remove many of the potentially eroding particles from the air before they reach the compressor. Both helicopter engines and turbo-fan engines in aircraft that fly low over the sea suffer from erosion due to the continual impact of salt crystals on the leading edges of fan and compressor blades.

Erosion rate is a function of the following.

- Operating environment;
- Component materials;
- Design;
- Protective coatings.

Predominately, erosion by inlet-airborne material is a compressor problem but it also affects other components in the airstream such as those in the by-pass duct. As particulates pass through the compressor, they are graded and the finer elements may be drawn into the cooling system. Not only does this process erode the internal air-washed parts, but it may lead to the glassification of sand particles in the turbine. When this happens molten silica solidifies on suitably 'cooler' surfaces, impedes cooling flows and may cause component failure due to overheating.

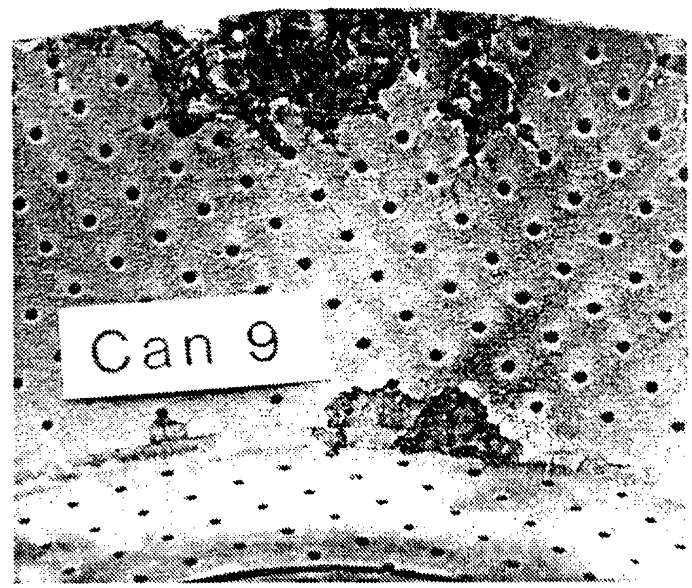



Figure 11 – Corrosion and spalling of thermal barrier coating in marine gas turbine caused by poor combustion

Particulates are frequently generated within the engine from:

- Erosion of abradable seal coatings by rotor blades;
- Hard carbon produced in the combustion section.

Although the critical components are not generally affected by the particulates carried in the mainstream flow, carry-through of hard particles into the cooling passages will remove material, particularly surface treatments, exposing the parent metal to corrosion.

## 2.7. FRETTING, GALLING AND WEAR

There are two sources of damage to contacting surfaces: fretting and wear. Fretting, shown magnified in figure 13, is due to oscillatory motion of very small amplitude, much smaller than may occur with normal wear. This can occur in joints that are bolted, keyed, press fitted, shrunk or riveted. Fretting also affects splines, couplings, clutches, spindles and seals. The effects of fretting are accelerated because the debris produced is trapped at the wear site and, whilst it may not lead to component failure, fretting produces small surface cracks which can propagate to a conventional fatigue failure. The start of such a crack is shown in figure 14.

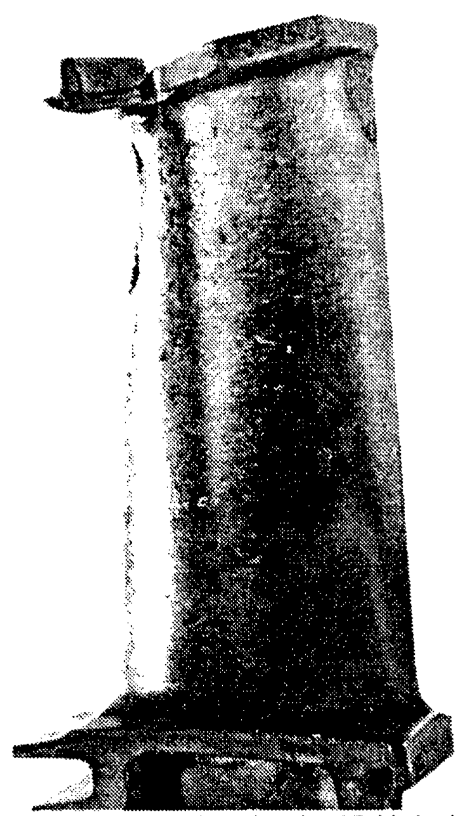


Figure 12 - Erosion of marine GT PT blade due to carbon in the gas stream from poor combustion

Wear will occur whenever two or more contacting components experience relative motion. Unlike fretting, wear is the result of large scale, unavoidable movement between two components. Typical examples of wear include a piston in the bore of an actuator, bearings, air and oil seals, and blade tips on abradable seals. The rate of wear is determined by a number of factors:

- Relative hardness of contacting materials. The softer material will wear more rapidly;
- Contact force. The higher the contact-force, the greater the wear rate;
- Lubrication. Lubrication is normally provided to control wear in components where relative motion is

intentional but areas where fretting occurs are rarely lubricated;

- Temperature. As temperature increases the wear rate tends to increase because materials become softer, lubrication becomes less effective and clearances tend to reduce, leading to an increased contact force.

The majority of fretting can be overcome by the application of anti-fretting compounds, elimination of the relative movement (often difficult), surface hardening, separation of the components or reduction of the applied stress. However, wear is inevitable between contacting surfaces in motion and must be controlled initially by design and then by maintenance.

Fretting or wear should not be confused with galling (or spalling). Galling is a fatigue mechanism. It is caused by cyclic loading, of a small area of the surface, of a component. It typically occurs in components such as bearings, where the passage of the rolling elements over the races induces a cyclic load. This can cause small cracks to grow into the surface, and then propagate parallel to the surface before re-emerging at the surface and releasing a small particle.

Fretting and galling are apparent on many blade/disk attachments. Galling (deep groves) and fretting (light groves) have a very different effect on fatigue life. Fretting tends to reduce HCF life and galling tends to reduce LCF life

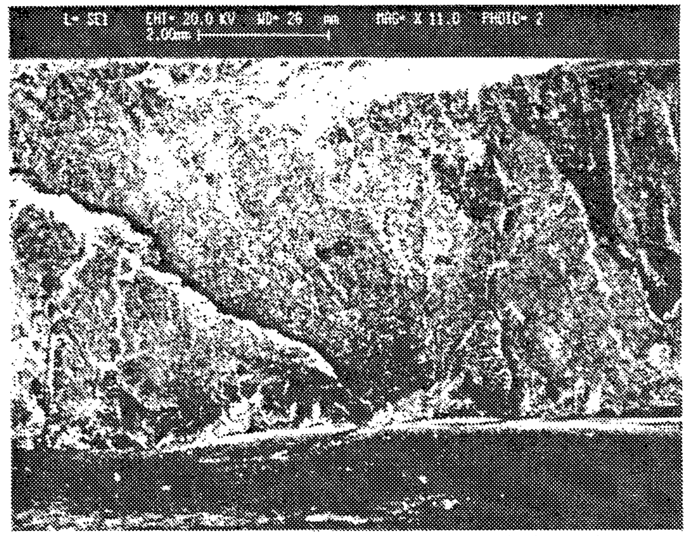


Figure 13 - Fretting in a titanium compressor disk dovetail

Figure 14 - Fretting crack with fatigue extension in training aircraft starter turbine

## 3. External Causes Of Life Consumption

### 3.1. Manufacturing And Material Defects

All metallic materials have finite fatigue lives and these are calculated based on the assumption that components will be manufactured to drawing and with material to the correct specification. However, in situations where the component does not comply with the drawing, the material is of incorrect composition, or heat treatment/surface conditioning has been incorrectly carried out then the achievable life can be greatly reduced. These issues are dealt with in more detail below:

#### 3.1.1. Components Not To Drawing

If a component is not to drawing, then a number of problems might arise. If the section of a component is too small then the stress for a given load will be increased. If a section is too thick it will be too stiff, its mass will be increased and, at a given operating condition, the loading that it exerts on other components might be increased. If a filet is of too small a radius then stress in that region will be increased or if the surface finish is too rough then cracks will be more likely to form early in the life of the component.

#### 3.1.2. Manufacturing Process Control

The component specification, which includes geometry, heat treatment, and hot and cold working, is vital in determining the service life of a component. A small alteration in the shape, composition, heat treatment cycle, or manufacturing process can have a major impact on life. Minor alterations in the chemical or mechanical composition of an alloy will alter its strength, fracture toughness and creep resistance.

Failure to heat treat at the required temperature or for the correct length of time may well result in a component that has the wrong grain structure or that has been inadequately stress relieved. Finally, a failure of the manufacturing process could lead to components having an incorrect metallurgical specification. For example, conditions required to create a single crystal turbine blade need to be very tightly controlled if the crystal structure is to be properly aligned to the span of the blade.

#### 3.1.3. Surface Conditioning

Surface conditioning includes all processes such as cold working and shot peening which alter the material

properties of the surface of the material. Most commonly, in gas turbine components, surface conditioning is used to impart a residual compressive stress into the surface of the component in areas where cracks are most likely to form. The effect of this is to inhibit crack initiation and thus extend life. If the chosen technique is not carried out sufficiently then the full benefit is not achieved. If it is over done then the surface can become excessively work hardened and brittle, or sub-surface tensile stresses may be generated, making it more prone to cracking.

#### 3.1.4. STRESS-RAISERS

Stress-raisers are features at the surface of, or embedded within, a component that cause stress contours to deviate from their normal (design) orientation, parallel to the direction of load, and to intensify around the tip of the feature. This is shown at figure 15. At the tip of the feature, in this case shown as a notch in the surface, there is a high stress concentration. The fatigue life will therefore be much reduced and a crack will start to propagate from this point.

Stress-raisers can be caused in a large number of ways.

- They can be present at manufacture as inclusions in the metal melt;
- Tools can leave scratches during machining;

Lack of care during transport, storage or handling.

### 3.2. BUILD OR MAINTENANCE ERRORS

One of the most effective ways to shorten the life of an engine right from the beginning is to build it incorrectly. Many mistakes can be made during the build or maintenance of an aero-engine. Some of these are listed below.

- Incorrect tolerances;
- Incorrect clearances;
- Incorrect torque loading;
- Insufficient cleanliness leading to blocked oil ways or dirt in bearings;
- Failure to apply lubricants;
- Incorrect assembly;
- Incorrect adjustment.

All of these can, and have, led to reduced engine life through early component failure or performance loss. All build and maintenance errors are avoidable but still occur regularly.

### 3.3. FOREIGN OBJECT DAMAGE

Foreign Object Damage (FOD) is an inevitable part of engine operations. Gas turbines have a high air mass-flow-rate and loose articles that are close to the intake of the engine are likely to be ingested. A gas turbine is able to lift objects from the ground in some circumstances. However, ingestion is much more likely to occur when a particle is already in motion. This motion can be due to a number of factors the most common of which are listed below.

- Jet efflux of another aircraft;
- Thrown up by aircraft wheels;
- High winds;
- Use of reverse thrust;
- Helicopter downwash;
- Inlet vortices.

There are other ways in which FOD can occur:

- Objects left in intakes during maintenance;
- Objects, particularly fasteners, from either the parent aircraft or an aircraft in close formation, typically during air-to-air refuelling;
- Bird ingestion;
- Ice ingestion.

Helicopters, due to their downwash and operation from unprepared sites, and combat aircraft, due to high specific thrust, low by-pass ratios and intake positioning, are worst affected by FOD.

Once damage is caused it will fall into one of these categories:

- Acceptable with no maintenance necessary;
- Acceptable but only after the damaged component has been blended or cropped to remove a stress raiser;
- Unacceptable, either the damaged blade will have to be changed in-situ, where this is possible, or the engine will have to be removed for repair;
- Catastrophic, leading to in-flight loss of an engine. This occasionally happens when aerodynamically induced blade flutter causes crack initiation and a HCF blade failure occurs.

Whatever the category of the FOD it may reduce the life of the engine. Even if acceptable, it will contribute to performance loss or cumulative damage and hence premature engine rejection.

#### 3.3.1. LIMIT EXCEEDANCE

Typical limit exceedances are listed below.

- Absolute Temperature. Any exceedance of a temperature limit will result in life being consumed at a greater than planned rate. An exceedance of an absolute temperature limit may result in damage to the turbine and/or combustion section.
- Time at Temperature. Most engines have temperature settings (ratings) that should only be used for limited periods. These include take-off, climb, combat, and emergency limits. The stress rupture life of the turbine blades will have been calculated on the assumption that these limits will not be exceeded. The longer the time spent at these elevated temperatures and hence rotational speeds (in the case of turbo-jets, turbo-fans and turbo-prop gas generators) the greater will be the rate of creep life consumption. It should be noted that current generation FADECs do not prevent time-at-temperature limits being exceeded.
- Rotational speeds. Whilst engines are designed to survive overspeeds of up to 122%, lifing calculations are based on speed limits being observed. Therefore, overspeeds result in increased dynamic loads that consume fatigue and creep life at a very high rate.

- Pressures. Over pressure results in increased loading on casings whose fatigue lives are therefore reduced.

The latest generation of digital engine-controls makes exceedance of engine operating limits unlikely. However, on older engines, particularly those with hydro-mechanical fuel controllers, limit exceedance is much more frequent. It is important that exceedances are recorded for subsequent maintenance action as defined by the manufacturer.

## 4. SUMMARY

The decision to apply a safe-life to an aero gas-turbine engine component is an expensive one. It is generally reserved for those components assessed as having failure modes that would hazard the integrity of the engine and the safety of the aircraft. These critical components typically include the major rotating assemblies and structural casings. The ability of engine components to resist failure is highly dependent on material properties that must be carefully selected to offer the optimum combination relevant to the engine type and application. Detail design considerations must also be taken to avoid stress concentrations and forcing functions that would promote failure.

Critical component life will be consumed in terms of low cycle fatigue, high cycle fatigue, thermomechanical fatigue and creep damage and it is essential to understand how the engine usage relates to the life consumption rate. Significant financial penalties are associated with excessively early retirement of critical components. Conversely, disastrous airworthiness consequences may be associated with late retirement. The situation is further complicated by other damage mechanisms which are common in gas-turbine usage and abusage, such as over-stress, corrosion, erosion, fretting, wear and impact damage, and which can reduce material properties and promote early failure of critical components.

During operation, there is little which can be done to increase component lives since the available life and the expected usage rate are inherent in the engine design. However, attention to component handling and assembly during maintenance, preservation and restoration of surface coatings, and avoidance of foreign object damage will all assist in achieving the published lives and avoiding expensive early failures.

## 5. REFERENCES

Photographs supplied by Dr G Harrison of DERA.

# Chapter 5
## Mechanics of Materials Failure

by

*(W. Beres)*

## 1. INTRODUCTION

This chapter develops the theme of chapter 4, and introduces analytical descriptions of the basic physics that describes the failure modes. Much of the content is further described in Appendix 3, and equations referred to as for example (A-20) may be found there.

## 2. MODELLING MATERIAL FAILURES

Let's define the terms *Failure* and *Failure Mode*:

- *Failure* of a gas turbine engine component is defined as any change in the size, shape, or mechanical properties of the component that makes it unable to satisfactorily perform its design functions. The term may also mean that all of the life is used up.
- *Failure mode* is defined as a physical or chemical process or processes, which take place or combine their effects to produce component failure.

| Engine Section | Component | Failure Mode |
|---|---|---|
| **Compressor** | **Blades** | **ER-COR, HCF** |
| | **Vanes** | **ER-COR, HCF** |
| | **Discs** | **LCF, C, HCF** |
| | **Spacer** | **LCF, C, HCF** |
| **Turbine** | **Blades** | **TF, C, HC, LCF, HCF** |
| | **Vanes** | **TF, HC, C, HCF** |
| | **Discs** | **LCF, C, HCF** |
| | **Torque Ring** | **LCF** |
| **Combustor Case** | | **LCF, C, HC** |
| **Shaft** | | **LCF, WR** |
| **Compressor discharge case (diffuser)** | | **ER, LCF,COR** |
| **Rotating seal** | | **LCF, C,HCF** |

| | | | |
|---|---|---|---|
| ER | Erosion | LCF | Low Cycle Fatigue |
| C | Creep | HCF | High Cycle Fatigue |
| COR | Corrosion | HC | Hot Corrosion |
| TMF | Thermo-mechanical Fatigue | WR | Wear |

Compressor disks and spacers in 1950-1970's engines are often steel and may therefore suffer corrosion, which reduces the LCF life.

Table 1 - Failure Modes and Life Limiting Properties for Turbine Engine Components

During long-term service in a corrosive environment under high stresses and at high temperatures, turbine engine components such as discs, blades or vanes suffer from cumulative damage. This gradually degrades their mechanical properties, and may lead to a component failure, or loss of engine structural integrity. Engine components can be subjected to both surface and internal damage.

- Surface damage due to erosion, corrosion, oxidation, wear, fretting or impact can promote crack nucleation at stress concentration areas. These cracks can then propagate under the influence of mechanical or thermal cyclic loads during service and eventually cause component failure.
- Internal damage is related to micro-structural changes that occur slowly at a rate which is strongly influenced by service temperature and stresses, which in turn depend greatly on user practice. These micro-structural changes can be associated partly with plastic damage accumulation and partly with metallurgical ageing. Depending on the component, alloy composition and service condition, the plastic damage may be due to creep, low cycle fatigue (LCF) or creep-fatigue interactions. As internal damage builds up, the resistance of components to deformation under static (creep) or cyclic (LCF) loading is reduced. Extensive loss of resistance to plastic deformation can lead to cavitation and internal cracking under creep conditions or ductility exhaustion and crack initiation under LCF or thermo-mechanical fatigue (TMF) conditions. Both lead to failure.

The primary failure modes that are assessed for life usage purposes are LCF and creep. A summary of all basic failure modes for gas turbine engine components is shown in Table 1.

LCF damage is dominant in bores, bolt hole and fillet areas of compressor and turbine discs. It is caused by stress cycling associated with engine start-up and shutdown and with engine speed excursions during service. While the driving force for LCF damage accumulation in compressor discs is primarily mechanical, in hot section areas of the engine it may also be associated with thermal cycling. Thermal fatigue damage in vanes and high-performance blades is often a life-limiting factor in modern gas turbine engines.

Creep damage is usually predominant in the mid-airfoil section of a turbine blade, and in some cases in the rims

of turbine discs, where the stresses and temperatures may be sufficiently high to cause time-dependent inelastic deformation.

Engine designers expend a great deal of effort to establish life limits for components whose deterioration may threaten operational safety. The lifing procedures employed to establish life limits vary with components and the predominant damage mode. In modern turbines, discs and spacers are normally designed to withstand LCF and burst due to overspeed as well as creep for the hot section side of the engine. Turbine blades and vanes are designed to withstand creep as well as thermo-mechanical fatigue and high-cycle fatigue (HCF).

The life limits for these components are initially established based on service estimates of damage accumulation rates. They are then revised as field experience is accumulated. Typical lifing procedures can be grouped as follows.

- For rotating components the most common lifing procedure employed to establish safe life limits follows a time or a cycle to crack initiation criterion. This is factored down to represent a minimum property component.
- For turbine blades and vanes, life limits is usually be prescribed by engine designers. In particular, turbine blades and vanes in aero-engines are seldom lifed because of the difficulties associated with predicting the service behaviour of metallurgically complex systems under conditions that can vary widely with user practice.
- In the case of coated components, these difficulties are compounded due to coating substrate inter-actions that are usually not well understood. A "life on-condition" approach is sometimes employed where, for instance, creep growth or untwist of airfoils is measured and distortion limits are used as retirement criteria.

As indicated, rotating parts such as compressor and turbine discs are usually life-limited due to LCF damage accumulation. The most common lifing, and traditional, method employed for these parts follows a "cycles-to-crack initiation" criterion. In this, a minimum life capability is defined statistically for simulated service conditions through extensive coupon testing and component test verification in a spin-pit test-rig. The statistical minimum is usually based on the probability that no more than 1 in 1000 components will have developed a detectable crack (typically chosen as a crack of 0.8 mm in length). It should be mentioned that the value '1 in 1000' is chosen arbitrarily. Different manufacturers use different values, e.g. 1 in 980, or 1 in 750 components.

Because many modern components have demonstrated high levels of resistance to crack growth, and modern materials have different characteristics, alternative lifing procedures based on damage tolerance and the application of fracture mechanics principles are now in use. The philosophy behind these alternative lifing procedures assumes that the component may be capable of continued safe operation during crack growth. This is subject to the proviso that the cracks grow sufficiently slowly during service to allow their growth to be reliably detected and perhaps even monitored through regularly scheduled inspections.

# 3. Stress States in Gas Turbine Engine Components

## 3.1. Thermal Analysis

The following three things must be known before the service life of aerospace components can be assessed:

- The environment and loads to which a component is subjected. This includes temperatures, flow rates, corrosive or erosive conditions, etc.
- The macroscopic response of the component to applied loads and environmental stresses and strains.
- The microscopic response of the component material to the local stress, temperature, and environment conditions.

The magnitudes, variations and exposure times for the stresses and temperatures experienced are the major factors controlling the lives of gas turbine engine components in service. The role of thermal and stress analysis is to calculate these quantities so that component service lives can be predicted. Particularly, application of damage tolerance based methodologies based on fracture mechanics (crack growth) principles requires detailed knowledge of stress fields in uncracked and cracked components. In addition, since superalloy materials applied in gas turbine engine components are sensitive to temperature, accurate determination of temperature conditions is very important.

Temperature and stress gradients across rotating turbine discs and spacers are dependent on conductive heat transfer within the component material and on convective and radiative heat flow at the component surfaces. Engine thermal analysis is usually performed using commercial or in-house finite-element or finite-difference based software. The associated internal air flow models for an engine take into account spool speeds, torque, shaft-loads, primary flow parameters (such as gas temperatures, mass flow, and pressures), and secondary flow analysis in cooling and pressure balancing air and oil flows.

## 3.2. Stress Analysis

Finite element based stress analysis procedures produce transient and steady state stresses in every location of a component taking into account external loads, such as the following.

- Centrifugal loads of discs and blades;
- Gas pressure generated loads;
- Torque in shafts;
- Thermal expansion stresses;
- Assembly loads, e.g. originating from bolt clamping.

Generally, full 3D finite element analysis of components is required, but for an axi-symmetric structure, there is no

dependency on one direction this allowing a so-called 2D analysis. 3D analysis is usually focused on the critical location or critical feature rather than the whole critical part. For example, analysis of bolted joints usually requires 3D finite element analysis. Body forces are calculated based on component mass and rotational speed. Temperature gradients that create thermal stresses in components are taken from the thermal analysis described in the previous section. Materials data are usually taken from comprehensive materials databases, which include data on changes of properties with temperature. In the main, linear static and dynamic calculations are performed, but for critical parts in critical locations temperature-dependent, non-linear-elastic, perfectly plastic behaviour of material is taken into account. In non-linear analysis, it is assumed that the structure responds in a linear elastic manner after stress redistribution due to plastic flow has been calculated within the component. Principal or von Mises stresses obtained from the calculation results are used to assess the integrity of a component and to identify fracture critical locations.

Potentially critical LCF locations can be identified by a relatively simple 2D analysis at steady state, which will identify high bulk-stress locations. These bulk stresses can then be multiplied by stress concentration factors to estimate peak stresses. The life-limiting feature on a particular critical part is most likely the one with the smallest complex geometry in the area of relatively high average stress. Identification of these critical features is one of the significant outputs of stress analysis. These features are then analysed for LCF life capability in terms of simple zero-max-zero cycles.

Combined with mission analysis the outputs of stress calculations may be used to describe stress as a function of time and identify the maximum stress points in the mission (critical mission points) and to identify critical and life limiting locations.

Stress results can be presented as 'snapshot in time' stresses on a part, feature or subsection of a feature, These represent the stresses at that specific point in the cycle. Results can be also presented as a function of time, as the stress variation in a specific period for a specific critical feature or a portion of that feature.

### 3.2.1. FINITE ELEMENT METHODS

Since in real life problems exact temperatures and stress solutions are rarely obtained analytically, numerical methods are used to obtain approximate solutions. Various numerical methods are used in engineering practice for solving boundary value problems in solid mechanics, including the finite element method, the boundary element method and the finite difference method. Of these methods, the finite element method is the most widely used.

The fundamental concept of the finite element method is that any continuous field variable, such as stress, pressure or temperature can be approximated by a discrete model. The discrete model is composed of a set of piecewise continuous sections, which are defined over a finite number of sub-domains of finite size known as elements. These elements are connected at specific points called nodes.

Variations of the unknown field variable inside the continuum are described by approximating functions, which are also known as shape functions. These functions are defined in terms of the values of the field variable at the nodal points. Field equations for every individual element are written and the set of unknowns representing the field values at the nodes is created. After assembly of the nodal values coming from different nodes, a set of linear and usually banded equations is created. Unknown nodal values of a field variable of interest are obtained by solving this system of linear equations. Usually, for stress analysis problems a set of displacements is obtained as a primary solution to the system of linear equations. Then, in the next step, the stresses are calculated from the displacements.

Usually the entire process of dividing the domain into finite elements, creating the system of equations, solving and interpreting the results is significantly simplified in widely used commercial finite element packages. The suitability of these should be assessed on a case-by-case basis for each task. Often aerospace companies use in-house finite element programs tailored for specific requirements such as rotor dynamics applications.

Typical finite elements used for thermal and mechanical stress analysis include:

- Axi-symmetric 3 and 4 node elements;
- Plane strain 3 and 4 node elements;
- 8 and 20 node bricks elements;
- 6 and 15 node wedge elements;
- 10 node tetrahedral elements;
- 2 node spring elements;
- 2 node gap elements.

Frequently, only segments of the rotating components such as discs and spacers are modelled to decrease the size of finite element models. Application of sliding boundary conditions on the cut out surfaces assures proper representation of displacement and stress fields in the disc segment.

### 3.2.2. FATIGUE AND FRACTURE ANALYSIS FOR LIFE PREDICTION

Results of the finite-element stress analysis are used to identify fracture critical locations in the components analysed.

The expected life is calculated, based on stress and temperature history in critical areas that are identified from the design mission. Cycles are commonly assessed with respect to their range and mean value, although dwell time may be an important factor for some materials that may undergo corrosion, oxidation or creep.

### 3.2.3. CRACK GROWTH

Crack growth prediction for components in service can be based upon various techniques with different degrees of sophistication and varying life prediction accuracy. In rising order of complexity these are:

- Rough estimates, where textbook stress intensity factor solutions are combined with stress predictions from finite element analysis;
- Analytical methods combined with the finite element method where tabulated stress concentration factors are used together with finite element calculations on coarse meshes;
- Sophisticated crack modelling in 3D, using automatic mesh-generation software, which remeshes the component as the calculated crack propagates. This is an iterative process, in which a 3D crack is modelled in the structure, the stress intensity distribution along the crack front is calculated using FEM, and a new crack front due to crack growth is made. The component is then remeshed and the stress calculation process starts again.
- Fracture mechanics analyses are performed mainly in using linear-elastic fracture-mechanics methods with a stress intensity factor as the critical parameter governing fatigue crack growth rate. Plastic non-linear analysis for fracture mechanics or life extension uses the contour integral called *J*-integral, Anderson (1991), Broek (1989), to correlate crack growth rates. Crack propagation laws, as described in Section 5.5.1, typically include range, temperature dependency, threshold effects and different R-ratio defined as a ratio of minimum to maximum stresses in a loading cycle.

### 3.2.4. STRESS INTENSITY FACTOR

The fracture behaviour of a given structure or material will depend on:

- Stress level;
- Material properties;
- The mechanism or mechanisms by which the fracture progresses.

The most successful approach in prediction and prevention of fracture has been to model the behaviour of the crack tip as simply as possible but including all significant parameters, such as:

- Crack length;
- State of stress;
- Fracture toughness.

The effects of temperature, environment, loading rate, and fluctuating loads must be taken into account by quantifying their influence on the basic parameters. To characterise the propagation of cracks in structures an energy-based approach and a stress-based approach are both used.

In an energy-based approach, unstable crack propagation is postulated to occur when the energy that could be supplied to the crack tip, during an incremental crack extension, is greater than or equal to the energy required for the crack to advance. The five energy components involved in an incremental crack extension are:

- Strain energy released;
- Energy supplied to the body by external work;
- Energy required to induce plastic deformation of the material at the crack tip;
- Kinetic energy;
- Energy required to create two new fracture surfaces.

The kinetic energy term and fracture-surface energy term are usually considered to be small and are neglected. Therefore, only the remaining first three energy components are considered in the analysis. A further simplification occurs if the size of the plastic zone at the tip of the crack is small compared to the total volume of the body. In this case, the strain energy released is a good approximation to the change in the elastic strain energy of a cracked body.

The stress intensity approach is a more general approach to characterise crack propagation. An essential feature of fracture mechanics is to characterise the local stress and deformation fields near a crack tip. An elementary fracture analysis of flawed components may be performed by stress analysis based on the theory of linear elasticity. The stress field surrounding a crack tip can be classified according to three major modes of loading which incorporate different crack surface displacements:

- Mode I - Opening mode, where the crack surfaces move directly apart in direction normal to the crack faces.
- Mode II - Sliding or in-plane shear mode, where the crack surfaces slide over one another in a direction perpendicular to the leading edge of the crack.
- Mode III - Tearing or anti-plane shear mode, where the crack surfaces move relative to one another and parallel to the leading edge of the crack. Mode III may occur when the component is subjected to shear or torsional loading.

In the majority of engineering configurations involving components with cracks, Mode I loading is experienced. Mixed mode I-II loading is sometimes encountered when a crack inclined to the load path exists.

The magnitude of the singular stress field is described, in linear-elastic fracture mechanics (LEFM), by the stress intensity factor *K*.. The unit for stress intensity factor is the product of the stress and the square root of length. This normally expressed as $Force/Length^{3/2}$, usually $MPam^{1/2}$. Because fatigue crack initiation is in general a surface phenomenon the stress-intensity factors for a surface crack, or a corner crack in a plate or at a hole, are solutions that are needed to analyse small-crack growth. For some more complex geometry and loading, SIF solutions for a large number of crack configurations have been generated and presented in handbooks, e.g. Rooke and Cartwright (1976), Tada et al. (1985), Murakami (1987). For very complex geometry and loading patterns, such as those experienced in gas turbine engine components, SIF solutions do not exist. In such cases, SIF should be estimated by numerical analysis.

The use of stress intensity factor to correlate fatigue crack growth is meaningful only when small-scale yielding conditions exist because plasticity and non-linear effects can change the stress distribution around a crack front

significantly. Among others, strain energy release rate and $J$-integral are quantities that describe a stress field at the crack tip when material plasticity is taken into account Anderson (1991), Broek (1982), (1989), Hertzberg (1983).

### 3.2.5. Stress Concentration Factor

The stress intensity factor described above is related to crack growth. It should not be confused with stress concentration factors. Stress concentration factors are applied to relatively crude stress calculations to take into account local features which disturb the local macro stress field.

# 4. Properties of Gas Turbine Engine Component Materials

This section describes the properties of materials used to manufacture gas turbine engine components. In particular, attention has been paid to the materials and the properties that most influence component in-service life.

Material properties can generally be divided into two main categories: mechanical (e.g. tensile, fatigue, creep) and physical (e.g. magnetic constant, thermal conductivity, thermal expansion coefficient). At low temperatures, mechanical properties are moderately affected by temperature changes, but time is of little importance. At high temperatures material behaviour becomes a function of both time and temperature. The properties of materials have been divided according to different criteria in the following sections.

## 4.1. Static Properties

### 4.1.1. Monotonic Stress-Strain Curve

The engineering tensile test is widely used to provide basic design information on the strength of materials. In this test, a specimen is subjected to a continually increasing uni-axial tensile force while the elongation of the specimen is recorded. Strain in this test is usually defined as *an average strain*. This is the ratio of the change in length of the specimen to its original length, $L$. Similarly, *an average stress* is the ratio of the tensile force to the initial cross sectional area of the specimen, $A$.

Since both the stress and the strain are obtained by dividing the load and elongation by constant factors, the load-elongation curve has the same shape as the engineering stress-strain curve. The main factors that influence the detailed shape and particular values on the stress-strain curve for a metal are metal composition, heat treatment, prior history of plastic deformation, strain rate, temperature and the state of stress applied during loading. It should be notes that strength increases with increasing strain rate. From a tensile test, the main material properties of interest: the *tensile strength*, *yield strength (yield point)*, *percent elongation* and *area reduction*, are derived, as shown in Appendix 3.

The stress-strain-curve and the flow and fracture properties derived from the tension test are strongly dependant on the temperature at which the test was conducted and on the applied strain rate used in the test. In general, strength decreases and ductility increases as the test temperature is increased. However, the general behaviour may be modified by microstructural changes.

## 4.2. Flow Properties

### 4.2.1. True-Stress-True-Strain Curve

The engineering stress-strain curve does not give a true indication of the deformation characteristics of a metal because it is based entirely on the original dimensions of the specimen, and these dimensions change continuously during the test. In addition, ductile metals pulled in tension become unstable and neck down during the test. Therefore, measures of stress, and strain based on the instantaneous dimensions are used. The *true stress* is the load at any instant divided by the cross sectional area over which it acts, while the *engineering stress*, or *conventional stress*, is the load divided by the original area.

Because during tension the cross sectional area of the specimen is decreasing sharply after necking starts, the load required to continue deformation decreases. Therefore, the engineering stress based on original area decreases after the point of maximum load has been reached. In reality the metal may continue to strain harden all the way up to fracture, so that the stress required to produce further deformation should also increase. If the true stress, based on the actual cross-sectional area of the specimen is used, it is obvious that the true stress-strain curve increases continuously up to fracture. If the strain measurement is also based on instantaneous strain, the curve is known as a *true-stress - true-strain curve*, or *flow curve*, since it represents the plastic-flow characteristics of the material. In the region of plastic deformation, the flow curve for metals may be expressed as a simple power law.

### 4.2.2. Effect of Strain Rate on Flow Properties

An increase in strain rate, defined as $\dot{\varepsilon}=d\varepsilon/dt$ increases the flow stress. This dependence is more pronounced with increases in temperature, as illustrated in figure 1, in which the plot is drawn in a double logarithmic scale. Strain rate sensitivity is a function of the base material and microstructure.

### 4.2.3. Effect of Temperature on Flow Properties

The stress-strain curve and the flow and fracture properties derived from the tension test are strongly dependent on the temperature at which the test was conducted. In general, strength decreases and ductility increases as the test temperature is increased. However, this general behaviour may be modified by structural changes, precipitation, strain ageing or recrystalisation, which may occur in certain temperature ranges. At high temperatures and/or long exposure times structural changes occur, resulting in time-dependent deformation or creep. The best way to compare the mechanical properties of different metals at various temperatures is to use the *homologous temperature*. This is the ratio of the test temperature to the melting point temperature expressed in Kelvin.

## 4.3. CREEP PROPERTIES

Creep is an important manifestation of inelastic time-dependent deformation behaviour. It is a time-dependent deformation of materials under stress at elevated temperatures, relative to their melting point. Creep behaviour is one of the most critical factors determining the integrity of elevated-temperature components, because creeping materials can slowly and continuously deform under constant stress over a period. Because of such deformation, unacceptable dimensional changes and distortions, ending in final rupture of the component, can occur. However, creep can have a positive effect on high temperature components because local creep phenomena can redistribute and thus reduce local stresses.

In its simplest form, creep is the progressive accumulation of plastic strain, in a component under stress at elevated temperature over a period of time. Creep failure occurs when the accumulated creep-strain results in a deformation of a component that exceeds the design limit. *Creep rupture*, used sometimes interchangeably with the term *stress rupture*, is an extension of the creep process to the limiting condition where the stressed component breaks. The interaction of creep with cyclic stressing and the fatigue process is of great importance in aircraft gas turbine technology.

Creep properties are generally determined by means of a test in which a constant uni-axial load or stress is applied to the specimen at constant temperature, and the resulting strain is recorded as a function of time. A creep curve is a plot of creep strain $\varepsilon_c$ against time $t$, but sometimes only time to rupture, $t_r$, is recorded during a creep test. A complete description of the creep behaviour of a metal necessitates measurements of several families of creep curves for various stress levels and temperatures.

The most general shape of a creep curve is shown in figure 2. This curve can conventionally be split up into three or four sections as shown in figure 3. It should be remembered that for a particular material under given conditions, not all creep stages are necessarily exhibited. For example, tertiary creep is predominant in super-alloys used in gas turbine engines. In figure 3, OA is the region of intial elastic deformation, which occurs as the full load is applied. It may be termed an incubation period, because it happens prior to the other creep stages. AB is called the primary creep stage. It is a period of decreasing creep rate where work hardening processes dominate. BC represents the secondary or steady state region of creep deformation. It is frequently the longest portion of a creep curve with equilibrium between strain-hardening and thermal softening. The final stage, CD, is termed the tertiary region. This period of accelerating creep rate occurs due to internal cavitation, and cracking and leads to complete fracture of the specimen.

A useful empirical model, which accurately represents the form of the creep curves obtained for most aero-engine materials and which applies over wide ranges of stress and temperature, is the equation developed in DERA, UK Harrison (1994),

$$\varepsilon_c = \sum C\sigma^{\beta} t^{\kappa} e^{Q/RT} \qquad (1)$$

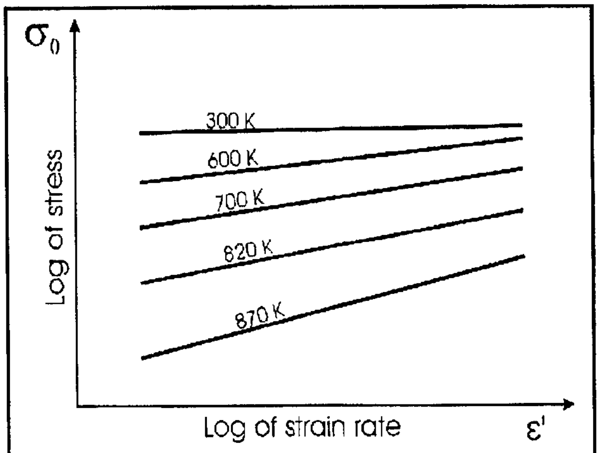


Figure 1 – Effect of temperature on strain rate

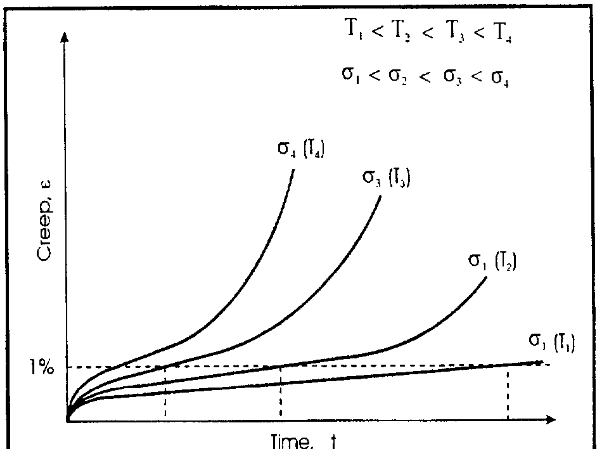


Figure 2 – Creep curves dependence on stress and temperature

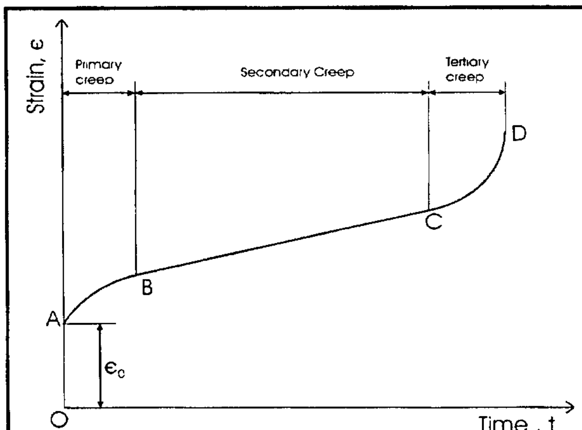


Figure 3 – Typical creep curve showing the three stages of creep

where $\varepsilon_c$ is the total creep strain, σ is the operating stress, t is time and T is the operating temperature, and the other values are constants. Although this set of equations is based on a phenomenological approach to creep modelling, the expression describes fully the primary, secondary and tertiary stages of the creep process. Of all the parameters pertaining to the creep curve, the most important for engineering applications are $\varepsilon_c$ and $t_r$,

specifically their dependence on temperature and applied stress. This depenedence varies with the applicable creep mechanisms.

Although for polycrystalline materials, the creep curve is conventionally divided into three regimes: primary, secondary and tertiary creep, in single crystal materials an 'incubation period', is often observed before the onset of the primary creep. This is really a period of initial creep acceleration but with extremely low inelastic strain. Depending on the material and the test conditions such as load, environment and temperature some forms of behaviour may be suppressed while another or others become predominant. For example, in single crystal super-alloys tertiary creep predominates the entire creep life while primary creep is negligible and secondary creep is not observed at all (Cary and Strudel, 1977 and 1978). When service exposed IN738, which is an investment cast polycrystalline material, is tested at 899°C and 90 MPa the entire creep curve is in the transient regime (Primary plus secondary creep). This is due to the effects of oxidation and precipitation of grain boundary carbides. Tertiary creep does not appear to occur. (Castillo, and Koul, 1988). To predict behaviour of a component in service and prevent the occurrence of the most dangerous mode, the engine designer and user must understand the physical mechanisms of creep and know their operating ranges of temperature and stress.

*Transient creep* is defined as the creep regime where the creep rate decreases with time until a minimum value, $\varepsilon_{min}$, is reached. For a polycrystalline material the creep rate, including the minimum creep rate, $\varepsilon_{min}$ , in this regime is often found to be sensitively dependent on the grain size and grain boundary micro-structural features, such as grain boundary serrations and grain boundary precipitate distribution. These observations indicate that grain boundary sliding (GBS) is the dominant deformation mechanism. GBS occurs as the result of dislocation glide along the grain boundary plane, which is more easily activated at the interfaces of micro-structural discontinuities, such as grain boundaries, upon application of a creep load (Langdon, 1970), (Wu and Koul, 1995). This mode of deformation can promote grain boundary cavitation and oxidation. The accumulation of this damage contributes to tertiary creep. In a single crystal material this mode of deformation/damage accumulation is absent, and transient creep is caused by dislocation multiplication and hardening due to the formation of a dislocation network.

*Tertiary creep* is defined as the creep regime where the creep rate increases with time, leading to the final rupture. The acceleration in creep rate may be caused by either or both dislocation multiplication (McLean, 1983), Dyson and Gibbons, 1987) or damage accumulation (Ashby and Dyson, 1985). Usually they happen together.

## 4.4. CYCLIC PROPERTIES

Any material subjected to repetitive or fluctuating stresses will fail at a stress much lower than that required for failure, on a single application of load in a tensile test. Failures occurring under such conditions are termed fatigue failures. Fatigue may be characterised as a progressive failure phenomenon that proceeds by the initiation and propagation of cracks. When the cracks are large enough the component becomes structurally unstable and catastrophic instantaneous failure occurs.

The damage done during the fatigue process is cumulative and generally unrecoverable. Vibrational stress on turbine blades, alternating bending loads on blades and fluctuating thermal stresses during starting and stopping or due to power changes are examples of cyclic loading that occur in gas turbine engine components.

Fatigue failure investigations over the years have led to the observation that the fatigue process includes two domains. One is of cyclic stressing and one of straining. These are significantly different in character, and in each of which failure is produced by different physical mechanisms. One domain of cyclic loading is that for which significant plastic strain occurs during each cycle. This domain is associated with high loads and low number of cycles to produce failure and is commonly referred to as low-cycle fatigue (LCF) or cyclic strain-controlled fatigue. The other domain of cyclic loading is that for which the strain cycles remains predominantly in the elastic range. This domain is associated with lower loads and high number of cycles to produce fatigue failure, and is commonly referred to as high cycle fatigue (HCF).

### 4.4.1. CYCLIC STRESS-STRAIN CURVE

Under strain controlled cyclic loading a material may remain cyclically stable or it may exhibit cyclic strain hardening or softening. The associated stress-strain hysteresis loops tend to stabilise after a relatively small number of cycles although as temperature increases, this 'shakedown condition' is correspondingly delayed and may never be reached. A cyclic stress-strain curve can then be constructed from peak shakedown stresses and corresponding strains, each LCF test providing a single pair of points. A power law similar to the curve for a monotonic stress-strain relationship may approximate the resulting cyclic stress-strain curve, see figure 4.

Cyclic *strain* controlled fatigue as opposed to our previous discussion of cyclic *stress* controlled fatigue, occurs when the strain amplitude is held constant during cyclic loading. LCF is found both in thermal cyclic fatigue, where a component expands and contracts in response to fluctuations in the operating temperature, and in reversed mechanical bending between fixed displacements.

The localised plastic strain in notches subjected to either cyclic stress or strain conditions results in strain controlled conditions near the root of the notch. This is due to the constraint effect of the larger surrounding mass of essentially elastically deformed material.

A metal may undergo cyclic strain hardening, cyclic strain softening or remain cyclically stable. This means that the material mass become either more or less resistant to the applied stresses or strains.

The mechanism of cyclic hardening and softening is related to the nature and stability of the dislocation substructure of the material. For an initially soft material,

the dislocation density is low. During plastic strain cycling the dislocation density increases significantly leading to significant strain hardening. For initially hard material, strain cycling causes a rearrangement of dislocations into a new configuration, which offers less resistance to deformation. This is strain softening. In both cases the newly generated dislocations assume a stable configuration for a given material and for a given magnitude of cyclic strain imposed.

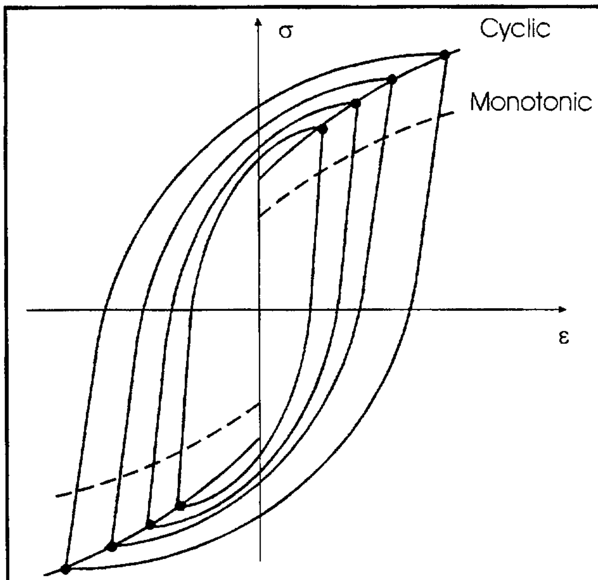



Figure 4 – Monotonic and cyclic stress-strain curves for a cyclically hardening metal

## 4.4.2. HIGH CYCLE FATIGUE (HCF)

The basic method for presenting HCF data is by means of S-N curves, figure 4, in which the number of cycles to failure is plotted against the applied stress range. $N$ is usually taken to denote the number of cycles applied at constant stress ratio, $R$, for complete specimen fracture. For some engineering materials, the S-N curve becomes almost flat at low stress, indicating a threshold value of stress below which failure will not occur for practical purposes. The threshold value of stress is defined as the fatigue limit. The S-N curve does not distinguish between crack initiation and crack propagation.

The number of cycles corresponding to the fatigue limit, i.e., at low stress, relates primarily to crack initiation, whereas for at high stresses the fatigue life corresponds primarily to crack propagation. Many variables in the test procedure, such as type of loading, mean stress, temperature, environment, specimen size, specimen surface condition, and stress concentrations, affect the fatigue life.

The *S-N* curve in the high-cycle region is sometimes described by the Basquin equation:

$$N\,\sigma_a^p = C \qquad (2)$$

where $N$ is the number of cycles to failure, $\sigma_a$ is the stress amplitude and $p$ and $C$ are empirical constants.

## 4.4.3. LOW CYCLE FATIGUE (LCF)

In gas turbine engine blades or discs, large mechanical stress and thermal gradients occur during operation. These may give rise to significant damage accumulation in only a few thousands of these large cycles during the design lifetime, so that LCF design and analysis methods are of great importance. For many components, the response of a material in the failure critical location was found to be strain dependent. It is assumed that smooth specimens tested under strain-control can simulate fatigue damage of the real components. This is because equivalent fatigue damage is assumed to occur in the material at the notch root, and in the smooth specimen, when both are subjected to identical stress-strain histories. In this fatigue domain the cyclic loads are relatively high, significant plastic strain is introduced during each cycle, and short lives or a low number of cycles to failure are exhibited.

The LCF test procedure results in the plot of the plastic strain range, $\Delta\varepsilon_p$, vs. the number of cycles, $N$, which is similar to the *S-N* plot. Figure 5 shows that a straight line is obtained when the test results are plotted on a double logarithmic scale:

$$\frac{\Delta\varepsilon_p}{2} = \varepsilon_f'(2N)^c \qquad (3)$$

where $\Delta\varepsilon_p/2$ is the plastic strain amplitude, $\varepsilon_f$ is the fatigue ductility coefficient defined by the strain intercept at $2N=1$, $2N$ is the number of strain reversals to failure (one cycle is two reversals), $c$ is the fatigue ductility exponent, which varies between -0.5 and -0.7 for many metals. This equation is known as the Coffin-Manson relation. A smaller value of $c$ results in a larger value of a fatigue life. The last equation describes the relation between the plastic strain component and fatigue life in the low-cycle-fatigue regime.

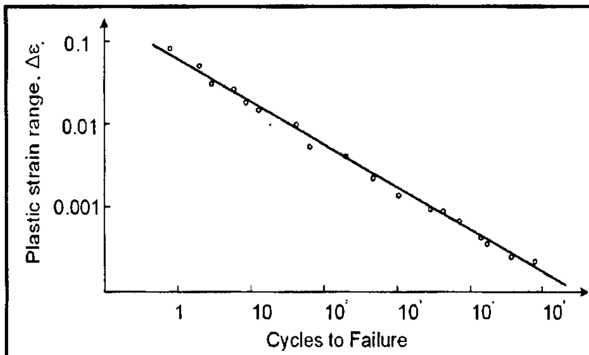



Figure 5 – Low cycle fatigue curve for steel

The elastic component of strain is often described in terms of a relationship between the true stress amplitude and the number of load reversals in a manner equivalent to the Basquin formula:

$$\frac{\Delta\varepsilon_e E}{2} = \sigma_a = \sigma_f'(2N_f)^b \qquad (4)$$

where $\Delta\varepsilon_e/2$ is the elastic stress amplitude, $E$ is the modulus of elasticity, $\sigma_a$ is the stress amplitude, $\sigma_f$ is the

fatigue strength coefficient, defined by the stress intercept at one load reversal $(2N_f=1)$, $N_f$ is the number of cycles to failure, $2N_f$ is the number of load reversals to failure, $b$ is the fatigue strength exponent, which varies in the range -0.5 and -0.12 for metals.

The fatigue resistance of a material subjected to a strain range could be estimated by superposition of elastic and plastic components. Summing Eq (3) and (4) results in the expression:

$$\frac{\Delta\varepsilon_t}{2}=\frac{\Delta\varepsilon_e}{2}+\frac{\Delta\varepsilon_p}{2}=\frac{\sigma_{f'}}{E}(2N_f)^b+\varepsilon_{f'}(2N_f)^c \tag{5}$$

The total strain life curve would approach the plastic strain life curve at large strain amplitudes and approach the elastic strain life curve at low total strain amplitudes, figure 6.

### 4.4.4. Thermo-Mechanical Fatigue (TMF)

Aircraft engine components, particularly hot section components, experience severe operating conditions involving complex combinations of cyclic mechanical stress, cyclic temperature and varying environmental conditions. Although the stress or strain that causes fatigue is often produced mechanically, it is perhaps even

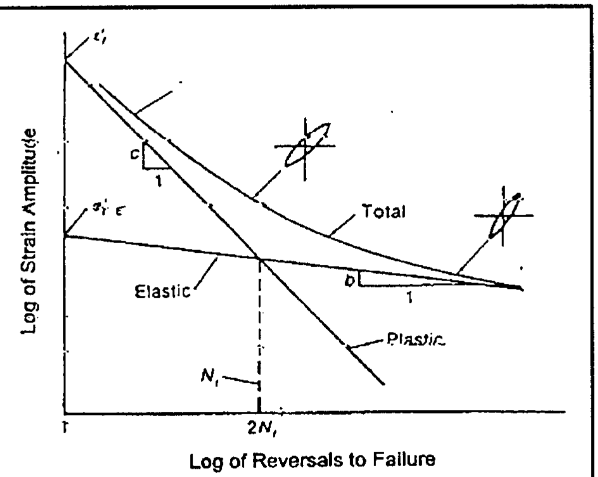



Figure 6 – Fatigue strain life curve obtained by superposition of elastic and plastic strain-life equations

more usual to find the cyclic strain produced by a cyclic thermal field. If the natural thermal expansions and contractions of a part are either wholly or partially constrained as often happens when a thermal gradient is applied to the component, then cyclic strains and stress result. These cyclic strains produce fatigue failure, just as if external mechanical loading produced them. Component cracking can be thus induced, not only mechanically but may be also caused or accelerated by thermally induced stress. The damage incurred by components under such conditions is known as *thermo-mechanical fatigue* (TMF). Thermal fatigue problems involve not only the complexities of mechanical loading, but all the temperature induced problems as well.

Most low cycle fatigue problems at high temperatures involve thermo-mechanical fatigue processes. Hot section components of gas turbine engines are subjected to cyclic temperatures simultaneously with cyclic stress. Analysis of these loadings and consideration of the associated fatigue damage becomes very complex, and many simplifications have to be introduced. Historically, thermal fatigue was considered as iso-thermal low-cycle-fatigue at the maximum temperature of the thermal cycle to which a component was subjected. However recent advances in test systems and numerical calculation methods have made it possible to conduct thermo-mechanical fatigue tests under well controlled conditions and also to analyse numerically complex thermo-mechanical cycles. It has been found that TMF loading can be more damaging than pure LCF loading for the same total strain range applied.

## 4.5. Fracture Mechanics Properties

### 4.5.1. Fatigue Crack Growth Rate (FCGR)

Fatigue cracks usually originate at geometric stress concentrations (holes and fillets) or at manufacturing flaws. The fatigue life of a component is divided into several phases: crack nucleation, micro-crack growth, macro-crack growth and failure. Crack nucleation is associated with cyclic slip and is controlled by the local stress or strain concentrations. Micro-crack growth, also referred to as small crack growth, is the growth of cracks from inclusions, voids or slip bands, in the range of 1 to 10 μm in length. Fracture mechanics parameters successfully correlate and predict fatigue crack growth and fracture in the region of macro-crack growth and failure. They can be applied once the crack has developed through several grains.

Reliable crack-propagation models permit the implementation of damage tolerant designs. These recognise the inevitability of cracks in engineering structures and aim at determining the critical crack load and length, which will preclude failure in a conservatively estimated service life.

Crack length increases with the number of cycles and the stress level. This may be expressed by a general plot of $da/dN$ versus a Stress Intensity Factor (SIF) range, $\Delta K$. The Fatigue Crack Growth Rate (FCGR) $da/dN$, is the crack extension $\Delta a$, during a small number of fatigue cycles $\Delta N$. FCGR is a function of the stress intensity factor range, $\Delta K$, the load ratio, $R$, temperature, and the environment.

Plots of $da/dN$ vs $\Delta K$ for a given $R$ are usually presented on double logarithmic graphs as shown in figure 7. This FCGR curve has a sigmoidal shape that divides the curve into three regions.

Region I is bounded by a threshold value $\Delta K_{th}$, below which there is no observable fatigue crack growth. At stresses below $\Delta K_{th}$ cracks behave as non-propagating cracks.

Region II is characterised by a linear log-log relationship between $da/dN$ and $\Delta K$. Crack growth rate in this region is influenced by material microstructures, mean stress, and environment. For a fixed load ratio, $R$, and a wide

range of $\Delta K$ values in region II, the data can be represented as a straight line on the double logarithmic plot. The Paris equation describes crack growth in region II as follows:

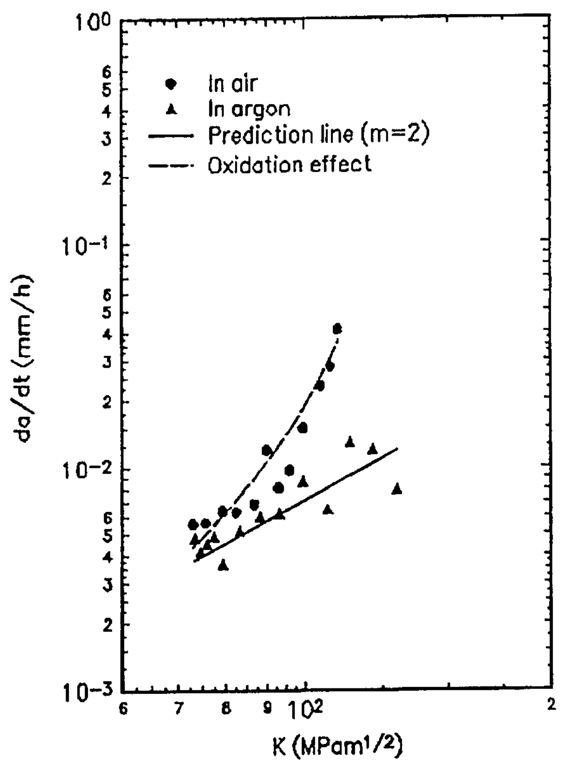



Figure 8 – Crack growth rate v. stress intensity factor for Udimet . After Wu and Koul (1996)

$$\frac{da}{dN} = C\left(\Delta K\right)^{n} \qquad (6)$$

where $C$ and $n$ are the empirical constants determined experimentally for a given material. The effects of load frequency, temperature, and operating environment are empirically contained in these constants. Note that in this formulation $C$ has the unit of $m/cycle(MPa\sqrt{m})^{-n}$ and $n$ is a dimensionless exponent which can be thought of as the slope of the fatigue crack growth rate curve. For most aerospace materials, the value of $n$ ranges from 2 to 4.

Special care should be taken if data from literature sources are used for component life prediction to ensure that the data corresponds to the actual environment. Most $da/dN$ vs $\Delta K$ curves are generated using $R \neq 0$ and room temperature. Interpolation of these curves to the actual conditions of intended application should be performed with great caution. Extrapolation beyond the testing region is not safe.

Eq. (6) provides a link between fracture mechanics and fatigue, because the fatigue life of the component can be obtained by the integration of this equation between the limits of initial crack size and the final crack size. This depends on knowing the stress intensity factor, $K$, for the cracked component as a function of component loading. Such integration usually has to be done by an iterative process.

Region III is a region of accelerating crack growth. Here $K_{max}$ approaches $K_c$, the fracture toughness of the material. In Region III, the fatigue crack growth rate rises to an infinite slope. FCGR is strongly influenced by microstructure, mean stress and specimen thickness in this region.

Increasing the mean stress in the fatigue cycle as described by a stress ratio causes an increase in the crack growth rates in all regions of the FCGR curve. Generally the effect of increasing $R$ is less in Region II than in Regions I and III. The Forman relation usually gives the influence of a stress ratio on the Paris formula:

$$\frac{da}{dN} = \frac{A(\Delta K)^{p}}{(1-R)K_c - \Delta K} \qquad (7)$$

where $K_c$ is the fracture toughness for the material and thickness, and $R$ is the stress ratio.

As mentioned before, the use of the stress intensity factor to correlate fatigue crack growth has meaning only when small-scale-yielding conditions exist at the crack tip. This is because plasticity and non-linear effects can change the stress distribution around the crack front significantly. A simple approach has been developed to modify the elastic stress-intensity factor to "correct" for plastic yielding at the crack tip. The approach was to add a plastic zone radius to the crack length and thus calculate stress intensity factor at the effective crack length ($a+r_y$). Other non-linear phenomena in crack growth behaviour include crack closure and the effect of constraints at the crack tip, but analysis of these effects definitely goes beyond the scope of this report.

The engineering problem of a crack existing in a structure can be partially solved by linear fracture mechanics. In particular, the following questions can be answered:

- What is the residual strength of a structure, as a function of crack size under a given load?
- What is the maximum permissible crack size that a structure can tolerate?
- How long does it take a crack to grow from its initial to the maximum permissible size?
- What is the service life of a structure when a certain flaw size is assumed to pre-exist?
- During the period available for crack detection, how often should a structure be inspected for cracks?

### 4.5.2. FRACTURE TOUGHNESS

The theory of Linear Elastic Fracture Mechanics (LEFM) provides a means for estimation of the fracture loads of structures containing sharp flaws of known size and location. An important observation from studying fracture behaviour is that the magnitude of the nominal applied stress, which causes fracture, is related to the size of the crack. For central through-the-thickness cracks, normal to the applied tensile stress, as the tensile loading on the pre-cracked plates is slowly increased, the crack length slowly increases and then abruptly extends to failure.

Experience has shown that the abrupt change from slow crack propagation to rapid crack propagation establishes an important material property named *fracture toughness*.

The fracture toughness is used as a design criterion in fracture prevention. It is defined as the critical stress intensity factor for material, $K_{Ic}$. While regarded as a material property, the fracture toughness may be strongly affected by operating temperature, heat treatment, and constraints. Therefore, care must be taken in design to use a value of fracture toughness that closely represents the expected operating temperature and environments for the actual component.

ASTM has developed a standard test procedure for determining plane strain fracture toughness. The compact tension specimen is the most commonly used test specimen geometry.

### 4.5.3. Creep Crack Growth Rate

Sustained load crack growth is time-dependant sub-critical crack growth occurring under a stress well below tensile failure. Creep crack growth usually occurs only at temperatures greater than 50% of the melting point and is not a failure process generally associated with fracture critical components.

To find the creep crack growth properties for the material, different cracked specimens are subjected to a constant load at elevated temperature and the crack extension is measured as a function of time. Examples of crack growth against time curves are shown in figure 8. These curves indicate an increasing crack propagation rate with increase in time after the crack growth has become well established. This is mainly because under constant load conditions, the magnitude of the stresses generated at the crack tip in the specimens increases with crack extension.

## 4.6. Influence of Mean Stress on Fatigue and Fracture

### 4.6.1. Influence of Mean Stress on HCF

Most of the fatigue data published have been determined for conditions of completely reversed stress cycles, $\sigma_m=0$. However, in engineering practice components are often loaded by superimposing alternating stress on a non-zero constant mean-stress.

There are several possible methods of determining and presenting an S-N curve where the mean stress is not equal to zero. The two most commonly used methods are presented in figures 9a and 9b. In figure 9a, the maximum stress is plotted against $N$ or $logN$ for constant values of stress ratio $R$. In the case of completely reversed stress $R$= -1.0. In figure 9b the alternating stress is plotted vs number of cycles to failure at a constant value of mean stress. Other methods of plotting these data include the maximum stress vs cycles to failure at constant mean stress and maximum stress vs cycles to failure at constant maximum stress.

### 4.6.2. Influence of Mean Stress on LCF

The Walker method used for correction for non-zero mean stress in the FCGR analysis is also applied as a way of deriving material LCF data for stress conditions other than zero minimum stress (A ≠1). The following formula is used:

$$\sigma_{(A\neq 1)} = \sigma_{(A=1)}\left[\frac{2R}{1-R}\right]^{(1-m)} \tag{8}$$

The Walker constant, $m$, having a value between 0 and 1 can be assumed to be a certain value based on a knowledge of the material. The Walker method is used to cover mean stress effects without generating full design curves for other mean stress conditions. The assumption may be conservative, i.e. it results in a higher derived stress than actual testing would show. It can be also verified by regression analysis of fatigue test data generated at various mean stresses, i.e. $R$ ratios. Application of this correction requires substantially less testing than would be necessary to generate a full design curve for the means stress conditions of interest.

## 4.7. Influence of Multi-Axial Stresses on Fatigue and Fracture

An effect of the multi-axial state of stress and strain fields on crack initiation and propagation is of paramount importance since most laboratory research involves tests on simple geometry specimens subjected only to either axial, torsional or bending loading. Such forces are rarely applied in combination in a laboratory environment. However in the real world complex cyclic forces varying in time may be imposed over a wide range of conditions. An example is that of thermal and mechanical stress transients in a gas turbine disc which create a bi-axial or tri-axial stress field. The stress in most engineering applications is bi-axial, while the state of strain is usually tri-axial. Miller and Brown (1984) and Ellyin (1997) presented a brief review of multi-axial fatigue problems.

A typical approach to account for the effect of multi-axiality on the LCF life, is to define *an equivalent total strain range*. This is calculated from the components of strain. An estimate of the life of a component subjected to multi-axial strain can be made from uniaxial low cycle fatigue data, expressed as total strain range vs number of cycles to failure.

## 4.8. Environmental Resistance

Environmental effects on the fatigue of metals may be severe. Quantitative fatigue-life predictions are often not possible because of the many environmental factors that influence fatigue behaviour, and lack of suitable data. For example:

- In corrosion fatigue, frequency effects can be quite substantial;
- In non-corrosive environments frequency is usually a second order concern;
- At elevated temperatures mean stress effects are extremely complex because of interactions between creep and fatigue, and the environment;
- At elevated temperatures, the linear elastic stress intensity factor, K, has more limitations because of appreciable plasticity;
- Substantial reductions in fracture toughness can occur at low temperatures, which reduce the critical crack sizes at fracture.

Design with environmental considerations places great

emphasis on real-life product testing, inspection, service history analysis, and experience.

### 4.8.1. OXIDATION RESISTANCE

Oxidation plays an important role in high temperature fatigue and creep. Protective oxide formation is a major factor in the fatigue resistance of a given material. These protective oxide films can be broken down by reversed slip, which causes much shorter high temperature crack initiation life. Crack propagation rates may also be accelerated by high temperature oxidation, and freshly exposed surfaces produced by local plasticity can oxidise rapidly.

Grain boundaries may be selectively attacked by oxygen. Tests at high temperature in a vacuum or inert atmosphere have shown substantial increases in fatigue and creep resistance compared to high temperature tests in air or combustion gases. Thus, local oxidation is a primary factor in the degradation of fatigue and creep resistance at high temperatures. Frequency effects are also substantially reduced at high temperatures in a vacuum or atmosphere.

### 4.8.2. CORROSION FATIGUE AND STRESS-CORROSION CRACKING

Many environments affect fatigue behaviour, and most components of gas turbine engines interact with air, water, salt water and hot gases. Corrosion fatigue refers to the joint interaction of a corrosive environment and repeated stressing. The combination of these two acting together is more detrimental than either acting separately. That is, repeated stressing accelerates the corrosive action and the corrosive action accelerates the mechanical fatigue mechanism. Corrosive environments may be detrimental even under static loads, particularly in higher strength alloys. Stress corrosion cracking is one form of environmentally assisted fracture that may occur after a sustained static load.

## 5. ANALYTICAL TECHNIQUES FOR DAMAGE CHARACTERISATION

### 5.1. CUMULATIVE FATIGUE DAMAGE

In gas turbine engine operation, the alternating stress amplitude varies during component service life, forming so-called spectrum loading. Therefore the direct use of standard S-N curves is very difficult in engine fatigue assessment because these curves were developed and presented for constant stress amplitude operation.. It is thus important to have a theory or hypothesis, verified by testing, that will permit good fatigue-life estimates to be made for components subjected to spectrum loading using standard S-N curves.

The basic postulate used for spectrum-loading fatigue evaluation is that any loading cycle at any given mean stress level produces fatigue damage. It is further postulated that the damage incurred is permanent and that operation of a component at several different stresses in sequence will result in an accumulation of total damage. When the total accumulated damage reaches a critical value, a fatigue failure occurs. This concept, however simple, is very difficult to implement in practice. The proper assessment of the amount of damage incurred by a component operating at any given stress level $S_i$ for a specified number of cycles $n_i$ is not easy. Many different cumulative fatigue damage theories have been proposed

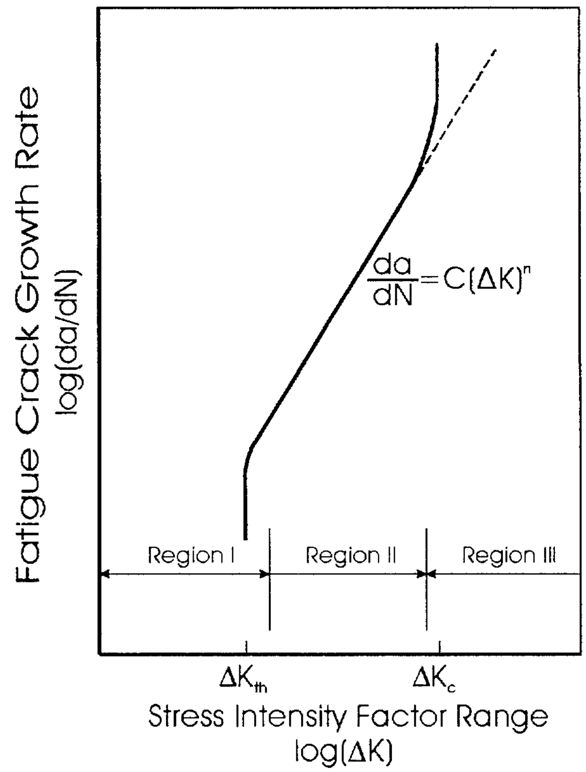


Figure 7 – Fatigue crack growth rate model

for assessing fatigue damage and fatigue failure caused by component spectrum loading. Of these theories, the Palmgren-Miner rule is presented here and a double linear damage rule is described in Appendix 3.

#### 5.1.1. PALMGREN-MINER RULE

One of the first cumulative fatigue damage theories was proposed by Palmgren (1924) and later developed by Miner (1945). This linear theory is generally referred to as the Palmgren-Miner rule or linear damage rule. It postulates that the damage fraction at any stress level Si, is linearly proportional to the ratio of number of cycles of operation at this stress level, ni, to the total number of cycles that would produce failure at that stress level, Ni:

$$D_i = \frac{n_i}{N_i} \tag{9}$$

Failure is predicted to occur if:

$$D = \sum_{j=1}^{i} \frac{n_j}{N_j} \geq 1 \tag{10}$$

Miner's rule simply sums the fractional life consumed by each type of damaging cycle identified, e.g. stop-max, idle-max. The number of cycles to crack initiation for each major and minor damaging cycle is determined by referring to the appropriate stress conditions on the minimum design S-N curves. The number of cycles or

hours required to accumulate enough cycles for the sum to reach 1 is defined as the life of the component for that type of cycle.

The main advantage of the Palmgren-Miner rule is it's simplicity. This rule, sometimes known as Miner's rule is widely accepted and used in the industry for LCF summation. It is simple, and fairly accurate when damaging minor cycles are of the same order of magnitude. It is best suited for LCF in engine components because there is not a great diversity of cycles. There are two significant shortcomings of these linear theories. The order of application of various stress levels is not taken into account. Damage is assumed to accumulate at a constant rate for a given stress level, regardless of component loading history.

## 5.2. Creep Damage

The design lifetimes of engineering components are often based on time to a specific strain or time to rupture. For design purposes it is usually more convenient to re-plot the data as shown in figures 10 and 11. Secondary creep-rate, time to specific strain (say 1%), or rupture life can then be read off at any desired stress and temperature. Data presented in figure 10 can be used for extrapolating to other stresses at a given temperature while data in figure11 can be used for extrapolating to other temperatures at a particular stress. Straight lines, with the slopes shown, will be obtained in the figures when the data can be correlated by the stress and temperature terms given in Equation (A-20) of Appendix 3.

A simple straight-line relationship is usually not obtained on plots like those in figures 10 and 11, when attempts are made to include a wide spread of data. This is because different values of n and Q are needed to describe different creep mechanisms. Consequently, to achieve more reliable extrapolations, time-temperature creep parameters have been devised for superimposing all of the results on the so-called 'master curve' for a material. The basis of the creep parameters is that time and temperature have similar effects. The same creep behaviour is obtained at the same stress in a short time at high temperature, as is attained in a long time at low temperature. This is clearly a simplification but it does result in satisfactory extrapolations of creep data.

The need for extrapolation techniques that permit estimation of the long-term creep and rupture strengths of materials based on short-duration tests is thus a very real and important one in design, quality control, and life prediction. Parametric techniques incorporate time, stress and temperature test data into a single expression. One of the parameters of concern is the time to rupture or time to specific strain.

Three parametric techniques are described in Appendix 3: the Larson-Miller parameters, the Monkman-Grant relationship, and the projection concept.

## 5.3. Creep-Fatigue Interaction

In life assessment of high-temperature components, the influence of interaction between failure modes has been considered. Hot section components in gas-turbine engines operate at high temperatures. Additionally, changes in conditions during engine starting, operation and shutdown result in large transient temperature gradients. If these transients are repeated, the differential thermal expansion during each transient results in thermally induced cyclic stresses. The extent of the resulting fatigue damage depends on the nature and frequency of the transients, the thermal gradients in the component and the material properties.

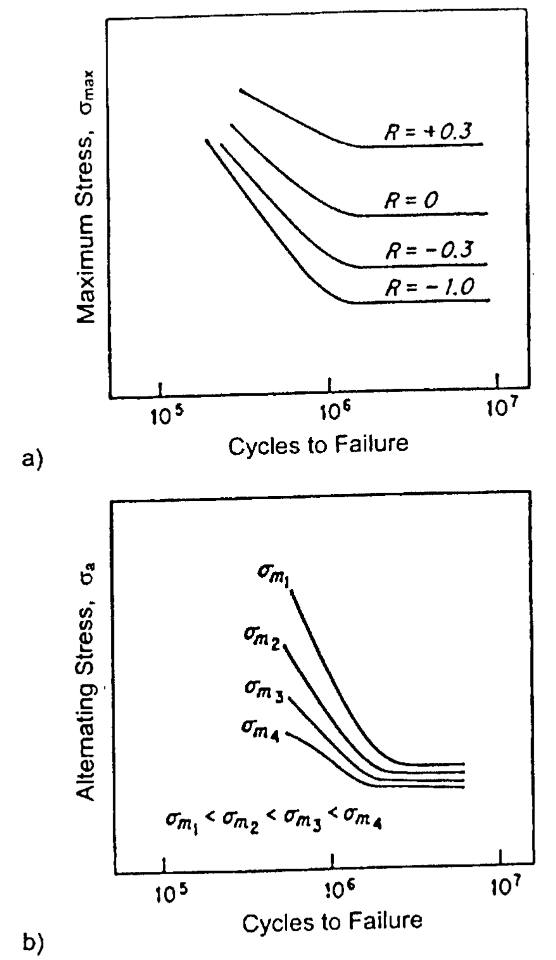



Figure 9 – Two methods of representing fatigue data for non-zero mean stress. After Dieter (1986)

Components that are subject to thermally induced stress generally operate within the creep range so that damage due to both fatigue and creep has to be taken into account. Gas turbine blades and disks are particularly subject to severe thermal gradients during start-up and shut down. During engine shut down, tensile stresses may be produced near cooling component surfaces, and may create thermal-shock cracks. In view of the importance of combined creep and fatigue damage with respect to component reliability, many attempts have been made to develop damage rules that will help in design and in component life prediction under creep-fatigue conditions.

To develop these damage rules, several types of laboratory test are used:

- Strain controlled tests with hold periods at constant strain;

- Creep tests under cyclic stress or strain;
- Alternating creep and fatigue tests;
- Strain-controlled tests under isothermal conditions;
- Strain-controlled tests under thermal and mechanical cycling.

The first test is the most common, while the last three tests are generally known as thermo-mechanical fatigue tests.

The principal method of studying creep-fatigue interaction has been to conduct strain-controlled fatigue tests with variable frequencies with and without a dwell period (hold time) during some portion of the test. The lower frequencies and the longer dwell periods can allow creep to take place. In pure fatigue tests, at higher frequencies and short dwell periods, the fatigue mode dominates and failures start near the surface and propagate trans-granularly. As the dwell period is increased, or the frequency decreased, the creep component begins to play a role with increasing creep-fatigue interaction. In this region, fractures are of a mixed mode, involving both fatigue cracking and creep cavitation. With prolonged dwell periods and occasional interspersed cycles, creep processes dominate, so that the damage mechanism can be treated as almost pure creep. In instances where oxidation effects contribute significantly to the creep fatigue interaction, the situation is more complex.

Several damage rules are used for estimating cumulative damage under creep-fatigue conditions. They include:

- The linear damage-summation method ,
- The strain range partitioning method ,
- The ductility exhaustion method.

These are described in more detail in Appendix 3.

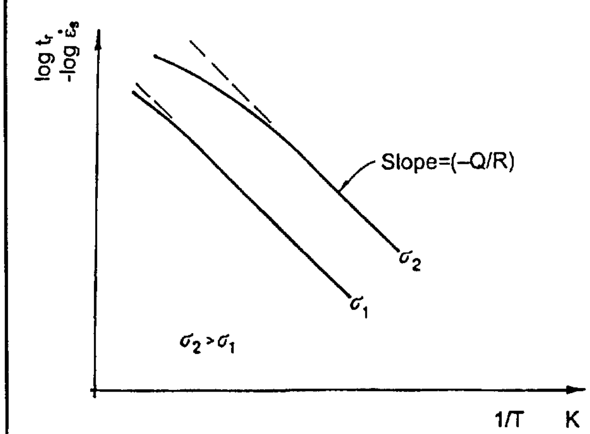



Figure 10 – Creep properties v. inverse of temperature at different stresses. After Webster and Ainsworth (1994)

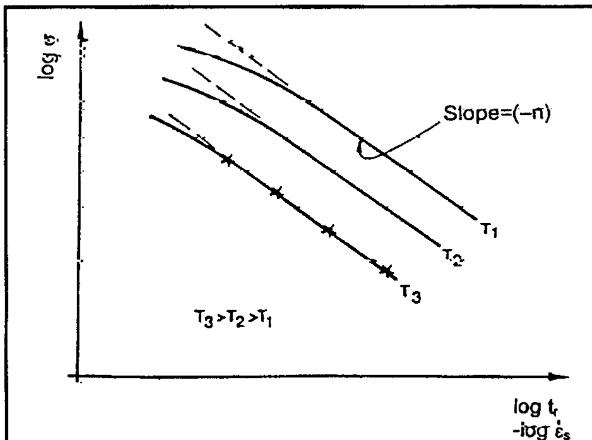



Figure 11 – Stress dependence of creep properties at different temperatures. After Webster and Ainsworth (1994)

## 6. Conclusion

A thorough understanding of the fundamentals of materials science is essential for practitioners in the field of usage monitoring. This chapter, together with appendix 3, seeks to provide a theoretical basis for further study. The following references provide a thorough supporting bibliography, and form the basis of this chapter.

## 7. References

*AGARD Engine Disc Co-operative Test Programme,* Ed. by A.J.A. Mom and M.D. Raizenne, AGARD SMP, AGARD-R-766, 1988.

*AGARD Engine Disc Co-operative Test Programme,* AGARD SMP, AGARD-R-766 (Addendum), 1993.

Agatanovic, P., and Heidenreich, R., *An Integrated Approach to Creep-Fatigue Life Prediction,* Low Cycle Fatigue, ASTM STP 942, pp. 500-518, 1988.

Anderson, T.L., *Fracture Mechanics*, CRC Press, 1991.

Annis, C.G., Jr., *An Engineering Approach to Cumulative Damage Fracture Mechanics in Gas Turbine Disks,* ASME Paper 79-GT-134, 1979.

Arnold, S.M. and Freed, A.D., *Constitutive Equations for Solid Mechanics*, Computational Structures Technology, Ed. by A.K. Noor and S.L. Venneri, ASME, pp. 1-38, 1995.

Ashby, M.F. and Dyson, B.F., *Creep Damage Mechanisms and Micromechanisms,* Advances in Fracture Research, Vol. 9, Pergamon Press, 1985.

Bammann, D. and Krieg, R.D., *Summary and Critique,* Unified Constitutive Equations for Creep and Plasticity, Ed. by A.K. Miller, Elsevier Applied Science, pp. 303-336, 1987.

Banks-Sills, L., *Application of the finite element method to linear elastic fracture mechanics,* Applied Mechanics Reviews, Vol. 44, No. 4, pp. 447-461, 1991.

Bannantine, J.A., Corner, J.J. and Handrock, J.L., *Fundamentals of Metal Fatigue Analysis,* Prentice Hall, 1990.

Barsom, J.M., *Structural Problems in Search of Fracture Mechanics Solutions,* Fracture Mechanics; Twenty-Third Symposium, ASTM, STP 1189, pp. 5-34, 1993.

Beres, W., *NSIFS Postprocessor for Displacement-Based Stress Intensity Factor Extractions from NISA II Calculations Result,* Report of the Institute for Aerospace Research, National Research Council Canada, Ottawa, ST-727, 1994.

Bernstein, H.L., *An Evaluation of Four Creep-Fatigue Models for a Nickel-Base Superalloy,* Low Cycle Fatigue

and Life Prediction, ASTM STP 770, pp. 105-134, 1982.
Bernstein, H.L., *at al., Prediction of Thermal-Mechanical Fatigue Life for Gas Turbine Blades in Electric Power Generation,* Thermomechanical Fatigue Behavior of Materials, ASTM STP 1186, pp. 212-238, 1993.
Besuner, P.M., *Probabilistic Fracture Mechanics,* Probabilistic Fracture Mechanics and Reliability, Martinus Nijhoff Publishers, pp. 387-459, 1987.
Bodner, S.R. *Review of Unified Elastic-Viscoplastic Theory,* Unified Constitutive Equations for Creep and Plasticity, Ed. by, Miller, A.K., Elsevier Applied Science, pp. 273-301, 1987.
Bogdanoff, J.L. and Kozin, F., *Probabilistic Models of Cumulative Damage*, John Wiley and Sons, 1985.
Boire-Lavigne, S., Gendron, S., Marchand, N.J. and Immarigeon, J.-P., *Modelling of Thermomechanical Fatigue Including Phase Transformations,* Thermal Mechanical Fatigue of Aircraft Engine Materials, AGARD SMP, AGARD-CP-569, pp. 17-1—17-10, 1996.
Brock, D., *Elementary Engineering Fracture Mechanics*, Martinus Nijhoff Publishers, 1982.
Broek, D., *The Practical Use of Fracture Mechanics*, Kluver Academic Publishers, 1989.
Broichhausen, K., *Effectiveness and Efficiency: The Challenge of Aero-engine Development*, XIII ISOABE Conference, Chattanooga, TN, 1997.
Burnett, D.S., *Finite Element Analysis*, Addison-Wesley, 1987.
Cary, C. and Strudel, J.L., *Apparent and Effective Creep Parameters in Single Crystals of Ni-based Superalloy—I. Incubation Period,* Acta Metallurgica, Vol. 25, pp. 767-777, 1977.
Cary, C., and Strudel, J.L., *Apparent and Effective Creep Parameters in Single Crystals of Ni-based Superalloy—II. Secondary Creep,* Acta Metallurgica, Vol. 26, pp. 859-870, 1978.
Castillo, R. and Koul, A.K., *Assessment of Creep Damage and Its Rejuvenation in Ni-based Superalloy*, Metallurgical Transactions, A, Vol. 19A, pp. 2049-2066, 1988.
Chaboche, J.L., *Lifetime Predictions and Cumulative Damage under High-Temperature Conditions,* Low Cycle Fatigue and Life Prediction, ASTM STP 770, pp. 81-104, 1982.
Chaboche, J.L., *Constitutive Equations for Cyclic Plasticity and Cyclic Viscoplasticity,* International Journal of Plasticity, Vol. 5, No. 3, pp. 247-302, 1989.
Chamis, C.C. and Johns, R.H., *Computational Engine Structural Analysis,* ASME Paper 86-GT-70, 1986.
Chulya, A. and Walker, K.P., *A New Uniformly Valid Asymptotic Integration Algorithm for Elasto-Plastic-Creep and Unified Viscoplastic Theories Including Continuum Damage,* NASA, TM-102344, 1989.
Coffin, L.F., *Some Perspectives on Future Directions in Low Cycle Fatigue,* Low Cycle Fatigue, ASTM STP 942, pp. 5-14, 1988.
Coffin, L.F., *Damage Monitoring, Life Prediction and Life Extension,* Low Cycle Fatigue and Elasto-Plastic Behaviour of Materials, Ed. by K.-T. Rie, Elsevier Applied Science, pp. 1-12, 1987.
Collins, J.A., *Failure of Materials in Mechanical Design*, John Wiley and Sons, 1993.
Cook, C.H., Spaeth, C.E., Hunter, D.T. and Hill, R.J., *Damage Tolerant Design of Turbine Engine Disks,* ASME Paper 82-GT-311, 1982.
Cook, T.S., Laflen, J.H., Van Stone, R.H. and Wright, P.K., *TMF Experience with Gas Turbine Engine Materials.* Thermal Mechanical Fatigue of Aircraft Engine Materials, AGARD SMP, AGARD-CP-569, pp. 16-1—16-12, 1996.
Cook, T.S. and Laflen, J.H., *Considerations for Damage Analysis of Gas Turbine Hot Section Components*, ASME Paper 84-PVP-77, 1984.
Cowles, B.A., *Life Prediction in High Temperature Environment: Overview of a Current Gas Turbine Engine Approach,* Materials Science and Engineering, Vol. A103, pp. 63-69, 1988.
Cowles, B.A., *High Cycle Fatigue in Aircraft Gas Turbines-An Industry Perspective,* International Journal of Fracture, Vol. 80, No. 2-3, pp. 147-163, 1996.
Dieter, G.E., *Mechanical Metallurgy, Third Edition,* McGraw-Hill, 1986.
Domas, P.A., *Elevated Temperature Component Fatigue Robustness-An Holistic Approach,* Thermal Mechanical Fatigue of Aircraft Engine Materials, AGARD SMP, AGARD-CP-569, pp. 21-1—21-13, 1996.
Donachie, M.J., Jr. and Sullivan, C.P., *Properties and Selection of Materials for Aircraft Gas Turbines,* Sawyer's Gas Turbine Engineering Handbook. Third Edition. Vol. III, Sawyer J.W. and Japikse, D. (Eds.), Turbomachinery International Publications, pp. 3-1—3-28, 1985.
Dundas, R., *Mechanical Design of the Gas Turbine,* Sawyer's Gas Turbine Engineering Handbook. Third Edition. Vol. I, Ed. by Sawyer J.W. and Japikse, D., Turbomachinery International Publications, pp. 10-1—10-88, 1985.
Dyson, B.F. and Gibbons, T.B., *Tertiary Creep in Ni-based Superalloy,* Acta Metallurgica, Vol. 35, pp. 2355-2369, 1987.
Edmunds, T.M., and Lawrence, R.A., *Monitoring Engine Thermal Stresses*, Engine Cyclic Durability by Analysis and Testing, AGARD PEP, AGARD-CP-368, pp. 2-1—2-20, 1984.
Ellyin, F., *Fatigue Damage, Crack Growth and Life Prediction*, Chapman and Hall, 1997.
Evans, W.J., Smith, M.E.F. and Williams, C.H.H., *Disc Fatigue Life Predictions for Gas Turbine Engines,* Engine Cyclic Durability by Analysis and Testing, AGARD PEP, AGARD-CP-368, pp. 11-1—11-13, 1984.
Fox, E.P. and Safie, F., Statistical Characterisation of Life Drivers for a Probabilistic Design Analysis, AIAA Paper 92-3414, 1992.
Fuchs, H.O. and Stephens, R.I., *Metal Fatigue in Engineering,* John Wiley & Sons, 1980.
Gan, X., Kilic, M. and Owen, J.M., *Flow and Heat Transfer Between Gas-Turbine Discs,* Heat Transfer and Cooling in Gas Turbines, AGARD PEP, AGARD-CP-527, pp. 25-1—25-11, 1995.
Halford, G.R., *Low Cycle Thermal Fatigue,* NASA TM-87225, 1986.
Halford, G.R., *Thermal Fatigue Durability for Advanced Propulsion Materials,* NASA TM-102348, 1989.
Halford, G.R., *Evolution of Creep-Fatigue Life Prediction Models*, Creep-Fatigue Interaction at High Temperature, ASME, AD-Vol. 21, pp. 43-57, 1991.

Halford, G.R., *Brief Summary of the Evolution of High-Temperature Creep-Fatigue Life Prediction Models for Crack Initiation,* Computational Methods for Failure Analysis and Life Prediction, NASA–CP–3230, pp. 121-150, 1993.

Halford, G.F., McGaw, M.A., Bill, R.C. and Fanti, P.D., *Bithermal Fatigue: A Link Between Isothermal and Thermomechanical Fatigue,* Low Cycle Fatigue, ASTM STP 942, pp. 625-637, 1988.

Halford, G.R., Meyer, T.G., Nelson, R.S., Nissley, D.M. and Swanson, G.A., *Fatigue Life Prediction Modelling for Turbine Hot Section Materials,* Toward Improved Durability in Advanced Aircraft Engine Hot Sections, ASME, IGTI—Vol. 2, pp. 97-107, 1988.

Halford, G.R., Saltsman, J.F., Verrilli, M.J. and Arya, V., *Application of a Thermal Fatigue Life Prediction Model to High-Temperature Aerospace Alloys B1900+Hf and Haynes 188,* Advances in Fatigue Lifetime Predictive Techniques, ASTM STP 1122, pp. 107-119, 1992.

Heine, J.E., Ruano, E. and DeLaneuville, R.E., *TMF Life Prediction Approach for Turbine Blades,* Thermal Mechanical Fatigue of Aircraft Engine Materials, AGARD SMP, AGARD–CP–569, pp. 20-1—20-6, 1996.

Harrison, G.F. and Edmunds, T.H., *The Role of Thermal and Stress Analyses in the Application of Damage Tolerant Design,* Damage Tolerance Concepts for Critical Engine Components, AGARD SMP, AGARD–CP–393, pp. 6-1—6-15, 1985.

Harrison, G.F. And Homewood, T. *The Application of the Graham and Walles Creep Equation to Aeroengine Superalloys*, Journal of Strain Analysis, Vol. 29, No. 3, pp. 177-184, 1994.

Harrison, G.F. and Shepherd, D.P., *Lifing Philosophies for Aero Engine Fracture Critical Parts,* Aero Engine Reliability, Integrity and Safety, The Royal Aeronautical Society/The Institution of Mechanical Engineers, pp. 5.1—5.12, 1991.

Harrison, G.F. and Winstone, M.R., *Modelling and Lifing of Structural Materials for Future Aeroengine Components,* Advanced Performance Materials, Vol. 3, No. 3/4, pp. 263-278, 1996.

Harrison, G.F., Tranter, P.H. and Williams, S.J., *Modelling of Thermomechanical Fatigue in Aero Engine Turbine Blades,* Thermal Mechanical Fatigue of Aircraft Engine Materials, AGARD SMP, AGARD–CP–569, pp. 2-1—2-15, 1996.

Hertzberg, R.W., *Deformation and Fracture Mechanics of Engineering Materials*, J. Wiley, 1983.

Hessler, U. and Domes, B., *LCF-Failure Analysis of an Aero-Engine Turbine Wheel,* Low Cycle Fatigue and Elasto-Plastic Behaviour of Materials, Ed. by K.-T. Rie, Elsevier Applied Science, pp. 664-670, 1987.

Hicks, M.A. and Pickard, A.C., *Life Prediction in Turbine Engines and the Role of Small Cracks,* Materials Science and Engineering, Vol. A103, pp. 43-48, 1988.

Kaufman, A. and Halford, G.F., *Engine Cyclic Durability by Analysis and Material Testing,* Engine Cyclic Durability by Analysis and Testing, AGARD PEP, AGARD–CP–368, pp. 12-1—12-12, 1984.

Köstner, A., Chataigner, E. and Rémy, L., *Thermal Fatigue, A Useful Tool to Assess Low Cycle Fatigue Damage in Superalloys for Components Experiencing Thermal Transient,* Thermal Mechanical Fatigue of Aircraft Engine Materials, AGARD SMP, AGARD–CP–569, pp. 8-1—8-8, 1996.

Koul, A.K., Wallace, W. and Thamburaj, R., *Problems and Possibilities for Life Extension in Gas Turbine Components,* Engine Cyclic Durability by Analysis and Testing, AGARD PEP, AGARD–CP–368, pp. 10-1—10-32, 1984.

Koul, A.K., Thamburaj, R., Raizenne, M.D., Wallace, W. and DeMalherbe, M.C., *Practical Experience with Damage Tolerance Based Life Extension of Turbine Engine Components,* Damage Tolerance Concepts for Critical Engine Components, AGARD SMP, AGARD–CP–393, pp. 23-1—23-22, 1985.

Koul, A.K., Immarigeon, J-P. and Wallace, W., *Emerging Technologies for Life-Cycle Management of Turbine Engine Components*, Canadian Aeronautics and Space Journal, Vol. 34, No. 1, pp. 27-37, 1988.

Koul, A.K., Bellinger, N.C. and Fahr, A., *Damage-Tolerance-Based Life Prediction of Aeroengine Compressor Spacers: I. A Deterministic Fracture Mechanics Approach,* International Journal of Fatigue, Vol. 12, No, 5, pp. 379-387, 1990.

Koul, A.K., Bellinger, N.C. and Gould, G., *Damage-Tolerance-Based Life Prediction of Aeroengine Compressor Spacers: II. A Probabilistic Fracture Mechanics Approach,* International Journal of Fatigue, Vol. 12, No, 5, pp. 388-396, 1990.

Kocks, U.F., *Constitutive Behavior Based on Crystal Plasticity,* Unified Constitutive Equations for Creep and Plasticity, Ed. by Miller, A.K., Elsevier Applied Science, pp. 1-88, 1987.

Laflen, J.R. and Cook, T.S., *Equivalent Damage – A Critical Assessment, Final Report*, NASA CR–167874, 1982.

Langdon, T.G. , *Grain Boundary Sliding in as a Deformation Mechanism During Creep,* Philosophical Magazine, Vol. 22, pp. 689-700, 1970.

Larson, F.R. and Miller, J. *Time-Temperature Relationships for Rupture and Creep Stresses*, ASME Transaction, Vol. 74, pp. 765, 1952.

Lasalmonie, A., Pellerin, F. and Fournier, D., *Thermomechanical Fatigue in Gas Turbine Engines: The Reason of a Concern,* Thermal Mechanical Fatigue of Aircraft Engine Materials, AGARD SMP, AGARD–CP–569, pp. 1-1—1-5, 1996.

Lemaitre, J. and Chaboche, J.-L., *Mechanics of Solid Materials,* Cambridge University Press, 1990.

Liebowitz, H., *Computational Fracture Mechanics*, Advances in Fracture Research, Eds. K.Salama, *et al.*, Vol. 3, pp. 1887-1921, 1989.

Lin, Y.K., and Yang, J.N., *On Statistical Moments of Fatigue Crack Propagation*, Engineering Fracture Mechanics, Vol. 18, pp. 243-256, 1983.

Malpertu, J.L. and Rémy, L., *Thermomechanical Fatigue Behavior of a Superalloy,* Low Cycle Fatigue, ASTM STP 942, pp. 657-671, 1988.

Manson, S.S., *Thermal Stress and Low-Cycle Fatigue*, McGraw-Hill, 1966.

Manson, S.S., Frecke, J.C. and Ensign, C.R., *Application of a Double Linear Damage Rule to Cumulative Fatigue*, Fatigue Crack Propagation, ASTM STP 415, p. 384, 1967.

Manson, S.S., *Future Directions for Low Cycle Fatigue,*

Low Cycle Fatigue, ASTM STP 942, pp. 15-39, 1988.

Manson, S.S. and Halford, G.R., *Re-examination of Cumulative Fatigue Damage Analysis—An Engineering Perspective,* NASA TM–87325, 1985.

Marscher, W.D., *Structural Design and Analysis of Modern Turbomachinery Systems,* Sawyer's Gas Turbine Engineering Handbook. Third Edition. Vol I., Sawyer J.W. and Japikse, D. (Eds.), Turbomachinery International Publications, pp. 7-1—7-56, 1985.

McClung, R.C., *et al., A Practical Methodology for Elastic-Plastic Fatigue Crack Growth,* Fracture Mechanics; 27th Volume, ASTM, STP 1296, pp.317-337, 1997.

McDowell, D., *Basic Issues in the Mechanics of High Cycle Metal Fatigue,* International Journal of Fracture, Vol. 80, No. 2-3, pp. 103-145, 1996.

McLean, B.F. and Dyson, B.F., *Particle Coarsening,* $\sigma_0$ *and Tertiary Creep,* Acta Metallurgica, Vol. 31 pp. 17-27, 1983.

McGaw, M.A. and Saltsman, J.F., *A High Temperature Fatigue Life Prediction Computer Code Based on the Total Strain Version of Strainrange Partitioning (SRP),* Creep-Fatigue Interaction at High Temperature, ASME, AD-Vol. 21, pp. 65-80, 1991.

McKnight, R.L., *Structural Analysis Applications,* Toward Improved Durability in Advanced Aircraft Engine Hot Sections, ASME, IGTI—Vol. 2, pp. 83-95, 1988.

Merkle, J., *Patterns and Perspectives in Applied Fracture Mechanics,* Fracture Mechanics; 26th Volume, ASTM, STP 1256, pp. 3-40, 1995.

Meyers, M. A. and Chawla, K.K., *Mechanical Metallurgy,* Prentice-Hall, 1984.

Miller, K.J. and Brown, M.W., *Multiaxial Fatigue: A Brief Revue,* Advances in Fracture Research (Fracture 84), Vol. 1, pp. 31-56, 1984.

Millwater, H.R., Wu, Y.-T., Cardinal, J.W. and Chell, G.G., *Application of Advanced Probabilistic Fracture Mechanics to Life Evaluation of Turbine Rotor Blade Attachments,* Journal of Engineering for Gas Turbines and Power, ASME, Vol. 118, No. 2, pp. 394-398, 1996.

Millwater, H.R., Chell, G.G. and Riha, D,S., *A Probabilistic Creep Fatigue Computer Program for Structural Integrity Analysis,* ASME Paper 96-GT-265, 1996.

Miner, M.A., *Cumulative Damage in Fatigue*, ASME Journal of Applied Mechanics, Vol. 67, pp. 159-164, 1945.

Monico, R.D. and Chew, J.W., *Modelling Thermal Behaviour of Turbomachinery Discs and Casings,* Heat Transfer and Cooling in Gas Turbines, AGARD PEP, AGARD-CP-527, pp. 24-1—24-10, 1992.

Moreno, V. and Jordan, E.H., *Prediction of Material Thermomechanical Response with a Unified Viscoplastic Constitutive Model,* International Journal of Plasticity, Vol 2, No. 3, pp. 223-245, 1986.

Murakami, Y (Ed.), *Stress Intensity Factors Handbook,* Pergamon Press, 1987.

Newman, J.C. Jr., *Advances in Fatigue Life Prediction Methodology for Metallic Materials,* Theoretical Concepts and Numerical Analysis of Fatigue, Eds. A.F. Blom and C.J. Beevers, pp. 301-325, 1992.

Newman, J.C., Jr., *The Merging of Fatigue and Fracture Mechanics Concepts: A Historical Perspective,* Fatigue and Fracture Mechanics: 28th Volume, ASTM STP 1321, pp. 3-151, 1997.

Newman, J.C., Jr., Booth, B.C., and Shivakumar, K.N. *An Elastic-Plastic Finite Element Analysis of the J-Resistance Curve Using a CTOD Criterion,* Fracture Mechanics: Eighteen Symposium, ASTM STP-945, D.T. Reed and D. Reed, Eds., ASTM, pp. 665-685, 1988.

Newman, J.C. Jr., Phillips, E.P., Swain, M.H. and Everett, R.A. Jr., *Fatigue Mechanics: An Assessment of a Unified Approach to Life Prediction,* Advances in Fatigue Lifetime Predictive Techniques, ASTM STP 1122, pp. 5-27, 1992.

Newlin, L., Sutharshana, S., Ebbeler, D., Moore, N, Fox, E., Annis, C. and Creager, M., *Probabilistic Low Cycle Fatigue Failure Analysis with Application to Liquid Propellant Rocket Engines,* AIAA Paper 90–1142–CP, 1990.

Noor, A.K., *Perspectives on Computational Structures Technology,* Computational Structures Technology, Ed. by A.K. Noor and S.L. Venneri, ASME, pp. 133-140, 1995.

Noor, A.H., and Pilkey, W.O., *State-of-the-Art Survays on Finite Element Technology,* ASME, 1983.

Palmgren, A. *Die Lebensdauer von Kugellagern,* Verfahrenstechnik, Vol. 68, pp. 339-341, 1924.

Penny, R.K. and Marriott, D.L., *Design for Creep,* Chapman & Hall, 1995.

Pickard, A.C., *The Application of 3-Dimensional Finite Element Methods to Fracture Mechanics and Fatigue Life Prediction,* EMAS, 1986.

Pickard, A.C., *Component Lifing,* Materials Science and Technology, Vol. 3, No. 9, pp. 743-749, 1987.

Pickard, A.C., Brown, C.W., and Hicks, M.A., *The Development of Advanced Specimen Testing and Analysis Techniques Applied to Fracture Mechanics Lifing of Gas Turbine Components,* International Conference on Advances in Life Prediction Methods, Ed. by D.A. Woodford and J.R. Whitehead, ASME, pp. 173-178, 1983.

Paris, P. C., *Reflections on Progress in Fracture Mechanics Research,* Fracture Mechanics: Twenty-Fourth Volume, ASTM, STP 1207, pp. 5-17, 1994.

Priest, R.H., Beauchamp, D.J. and Ellison, E.G., *Damage During Creep-Fatigue,* International Conference on Advances in Life Prediction Methods, Ed. by D.A. Woodford and J.R. Whitehead, ASME, pp. 115-122, 1983.

Romanoski, G.R., Antolovich, S.D. and Pelloux, R.M., *A Model for Life Predictions of Nickel-Base Superalloys in High Temperature Low Cycle Fatigue,* Low Cycle Fatigue, ASTM STP 942, pp. 456-469, 1988.

Rémy, L., Rezai-Aria, F., Danzer, R. and Hoffelner, W., *Evaluation of Life Prediction Methods in High Temperature Fatigue,* Low Cycle Fatigue, ASTM STP 942, pp. 1115-1132, 1988.

Riedel, H., *Creep Crack Growth*, Fracture Mechanics: Perspectives and Directions (Twentieth Symposium), ASTM STP 1020, pp. 101-126, 1989.

Robinson, E.L., *Effect of Temperature Variation on Long-Time Rupture Strength of Steels,* ASME Transactions, Vol. 74, pp. 777-801, 1952.

Rooke, D.P., and Cartwright, D.J., *Compendium of Stress*

*Intensity Factors*, Her Majesty's Stationery Office, London, 1976.

Schubert, F., Penkalla, H.J. and Frank. D., *Thermal Mechanical Fatigue of Model Blades Made from CC and DS Superalloys,* Thermal Mechanical Fatigue of Aircraft Engine Materials, AGARD SMP, AGARD–CP–569, pp. 22-1—22-9, 1996.

Sehitoglu, H., *Thermo-Mechanical Fatigue Life Prediction Methods,* Advances in Fatigue Lifetime Predictive Techniques, ASTM STP 1122, pp. 47-76, 1992.

Shih, C.F, deLorenzi, H.G. and German, M.D., *Crack Extension Modelling with Singular Quadratic Isoparametric Elements,* International Journal of Fracture, Vol. 12, pp. 647-651, 1976.

Slavik, D. and Cook, T.S., *A Unified Constitutive Model for Superalloys,* International Journal of Plasticity, Vol 6, No. 7, pp. 651-654, 1990.

Sokolowski, D.E. and Ensign, C.R., *Toward Improved Durability in Advanced Combustors and Turbines—Progress in the Prediction of Thermomechanical Loads,* ASME Paper 86-GT-172, 1986.

*Sawyer's Gas Turbine Engineering Handbook. Third Edition. Vol I.,* Ed. by J.W. Sawyer and D. Japikse, Turbomachinery International Publications, 1985.

Saxena, A., *Application of Fracture Mechanics in Maintenance of High Temperature Equipment - An Assessment of Critical Needs*, Fatigue and Fracture Mechanics: 28th Volume, ASTM, STP 1321, pp. 70-85, 1997.

Sih, G.C., *Fracture Mechanics in Two Decades*, Fracture Mechanics: Perspectives and Directions (Twentieth Symposium), ASTM STP 1020, pp. 9-28, 1989.

Stouffer, D.C. and Salemme, G., *The Impact of Constitutive Modeling on Life Prediction,* Fatigue and Fracture of Aerospace Structural Materials, ASME, AD–36, pp. 163-176, 1993.

Sutharshana, S., Creager, M., Ebbeler, D. and Moore, N, *A Probabilistic Fracture Mechanics Approach for Structural Reliability Assessment of Space Flight Systems*, Advances in Fatigue Lifetime Predictive Techniques, ASTM STP 1122, pp. 234-246, 1992.

Sutharshana, S., Newlin, L., Ebbeler, D., Moore, O'Hara, K., E. and Creager, M. *Probabilistic High Cycle Fatigue Failure Analysis with Application to Liquid Propellant Rocket Engines,* AIAA Paper 90–1141–CP, 1990.

Tada, H., Paris, P.C, and Irwin, G.R., *The Stress Analysis of Cracks Handbook*, Del Research Corporation, Bethlehem, PA, 1985.

Tanaka, K., *Mechanics and Micromechanics of Fatigue Crack Propagation*, Fracture Mechanics: Perspectives and Directions (Twentieth Symposium), ASTM STP 1020, pp. 151-183, 1989.

*Test Cases for Engine Life Assessment Technology,* AGARD PEP, AGARD–AR–308, 1992.

Thamburaj, R., Koul, A.K., Wallace, W. and DeMalherbe, M.C., *Life Extension Concepts for Canadian Forces Aircraft Engine Components: J85-CAN-40 Compressor and Turbine Material*, National Research Council Canada, National Aeronautical Establishment Report, LTR-ST-1585, 1986.

Van Stone, R.H., *Residual Life Prediction Methods for Gas Turbine Components,* Materials Science and Engineering, Vol. A103, pp. 49-61, 1988.

Vormwald, M., Heuler, P. and Seeger, T. *A Fracture Mechanics Based Model for Cumulative Damage Assessment as Part of Fatigue Life Prediction*, Advances in Fatigue Lifetime Predictive Techniques, ASTM STP 1122, pp. 28-43, 1992.

Viswanathan, R., *Damage Mechanisms and Life Assessment for High-Temperature Components*, ASM International, 1989.

Viswanathan, R., *Low Cycle Fatigue Life Prediction of Fossil Plant Components,* Low Cycle Fatigue and Elasto-Plastic Behaviour of Materials–3, Ed. by K.-T. Rie, Elsevier Applied Science, pp. 1-12, 1992.

Walker, K. *The Effect of Stress Ratio During Crack Propagation and Fatigue for 2024 T3 an 7075 T6 Aluminium,* Effects of Environment and Complex Loading History on Fatigue Life, ASTM, STP-462, pp. 1-??, 1970.

Walker, K.P., *Research and Development Program for Nonlinear Structural Modeling with Advanced Time-Temperature Dependent Constitutive Equations,* NASA, CR–165533, 1981.

Walker, K.P. and Freed, A.D., *Asymptotic Integration Algorithms for Nonhomogeneous, Nonlinear, First Order, Ordinary Differential Equations,* NASA, TM–103793, 1991.

Webster, G.A., Ainsworth, R.A., *High Temperature Component Life Assessment*, Chapman and Hall, 1994.

Wei, R., *Life Prediction: A Case for Multidisciplinary Research,* Fracture Mechanics; 27th Volume, ASTM, STP 1296, pp. 3-24, 1997.

Wu, X.J., Wallace, W., and Koul, A.K., *A New Approach to Fatigue Threshold*, Fatigue and Fracture of Engineering Materials and Structures, Vol. 18, No. 7/8, pp. 833-845. 1995.

Wu, X.J. and Koul A.K., *Modeling creep in complex engineering alloys,* Creep and Stress Relaxation in Miniature Structures and Composites, Ed. by H.D. Merchand, TMS Publications, pp. 1-19, 1996.

Wu, X.J. and Koul, A.K., *Grain Boundary Sliding in the Presence of Grain Boundary Precipites*, Metallurgical Transactions, A, 26A, pp. 905-914, 1995.

Yamauchi, M., *at al., Comparison of Creep-Fatigue Life Prediction by Life Fraction Rule and Strainrange Partitioning Methods,* Low Cycle Fatigue, ASTM STP 942, pp. 399-413, 1988.

Zienkiewicz, O.C., *The Finite Element Method*, McGraw-Hill, 1977.

# Chapter 6

## Translation of Service Usage into Component Life Consumption

by

*(G. Harrison)*

# 1. INTRODUCTION

This chapter discusses the techniques used to determine the structural component life with the safety levels imposed by the regulatory authorities. It describes how materials algorithms may be developed and used, both to establish the declared component life and to evaluate the rate at which this life is consumed under service conditions.

Certification is a process that airborne equipment has to pass through to demonstrate that it meets the safety regulations specified by the various airworthiness authorities. Typical examples of certification requirements include:

- Fan blade-off tests where an explosive mixture is used to cause the release of the whole of a single airfoil and root fixing. The objective is to show that in the extremely unlikely event of a whole blade being released, the fan casing would contain the blade.
- Cutting the shaft that takes the drive from the power turbine, to demonstrate that the control system can shut down quickly enough to prevent the loss of turbine blades, or a disk burst, during the resultant over-speed.

Whether such tests are passed or failed is immediately apparent.

In the case of aero-engine fracture critical components exposed to widely varying thermo-mechanical loading conditions, the failure modes progress in a complex manner. Two of the more important variables that influence these processes are the material that an individual component is made from, and the way in which the engine is used.

At the end of the engine development process, each fracture critical part is 'qualified' to the extent that it has passed a set of tests, which are designed to demonstrate that the regulatory requirements have been met. There may be some restrictions or limitations on the operational use, but these should not be sufficient to prevent flying. At this point an appropriate service life is authorised or recommended, depending on national practice, by the design authority for each safety critical structural component. In many countries, the authorising body is the engineering branch of the military service that operates the aircraft. When a component reaches the 'declared' life it is withdrawn from use.

The national qualification requirements are gradually being harmonised. In respect of the manufacturers' approaches towards their fulfilment, we have attempted to describe the broad concepts adopted and to illustrate typical techniques in general use.

# 2. SAFETY REQUIREMENT

Regardless of the details of the lifing methodology, it is essential that all declared lives be to a common level of safety. For many civil and military engines, service lives of fracture critical components are specified such that with a confidence level of 95%, not more than 1/750 (or 1/1000) will achieve the defined failure point. For damage tolerance lifed components two types of flaw must be considered. For surface based flaws the safety criterion may be set to safety levels of 1 in 750. However, for buried flaws the criterion may be set such that not more than 1 in 10,000 components will exceed the defined failure point. This is because buried flaws are difficult to detect.

# 3. DEFINING THE FAILURE POINT

## 3.1. LIFE TO FIRST CRACK

Within a general safe-life philosophy there are several definitions associated with 'failure', where failure is the defined end-point. The Life-To-First-Crack definition uses a crack depth of 0.38 mm as the criterion for component failure. This means that all discs should be removed from service at the stage at which it is calculated that a crack of 0.38 mm. depth would be present in the weakest member of the total service population. Each disk is therefore treated as though *it* is the weakest disk in the fleet.

As stated above, a typical requirement is that with a confidence level of 95%, not more than 1-in-750 components will exceed the declared failure point, defined as a 0.38 mm deep crack.

Whilst this approach has successfully ensured that service failures of engine disks are extremely rare, a natural consequence is that most disks are retired from service after consuming just a small fraction of their available life. Indeed, the average disk has consumed only 40% of its available Life-To-First-Crack, (LTFC).

The depth of 0.38 mm was originally based on the effectiveness of crack detection techniques. Modern methods can easily exceed this, but the definition provides a standard. It is emphasised that beyond 'first crack' the remaining (crack growth) life of the disc is a significant portion of the cycles to dysfunction. In many components, this portion may be typically 50% of the declared safe life. However, minor cycles are relatively more damaging in propagation than in initiation. Hence, in terms of the remaining additional missions, or engine flying hours that can be flown prior to disc burst, this can be less than 20% of the initiation life.

## 3.2. 2/3 DYSFUNCTION-LIFE AND DAMAGE TOLERANCE

In Europe, many newer components are designed so that the failure point is based on 2/3 of the dysfunction life. This is a broader concept that allows failure criteria other than surface cracks of a defined size to be taken into consideration. In general, lives declared using both methods are broadly similar, but the latter approach provides a determined safety level. For damage tolerant designs in which the 2/3 dysfunction-life exceeds the life to a 0.38-mm deep crack, this approach provides the opportunity for safe life extension.

### 3.2.1. DATABASE APPROACH

In the database approach, a fracture mechanics method is used to combine both specimen and full-scale component

tests on different design features into a common database for the material. If the respective stress intensities can be calculated for all relevant cases, the back-calculation of effective initial-defect sizes for all fatigue results together with a statistical analysis of the size distribution, allows a maximum probable-flaw size to be established. It is a simple process to use this effective initial-flaw size in a crack growth calculation to enable minimum component lives to be established for each design feature. Declared service lives are based on either the 0.38-mm life LTFC, or the 2/3 dysfunction-life.

In the above context, an assessment of a wide range of current aero-engine disc designs has revealed that in most cases, the application of the LTFC and the 2/3 dysfunction-life concepts give very similar values for declared lives. In general, the constant safety margin of the 2/3 dysfunction-life criterion is preferred. This is because the use of higher strength materials at increased operating stresses has the potential effect of reducing the critical crack size, associated with rapid crack growth, to below the 0.38 mm depth of the engineering 'first-crack'.

In European practice, where it can be shown that the 2/3 dysfunction-life exceeds the identified LTFC, fracture mechanics crack propagation methods may be used to determine the available service life beyond 'first crack'. For potential sub-surface locations, such as the disc mid-cob region, probabilistic crack growth methods are used to account for the effects of defects of specific sizes, at specific locations. As discussed below, fracture mechanics methods may also be used uniquely from an established NDE size.

## 3.3. DAMAGE TOLERANCE

In the US ENSIP (Ref. Mil Std 1793 (USAF), (1984)) approach for defect tolerant designs, declared lives are based on quantifying the available safe propagation life of fatigue cracks. The initial crack size used in fracture mechanics based lifing calculations, is defined as the maximum defect size that is associated with a detection probability of 90% to a confidence level of 95%. This relates specifically to the equipment used in the NDI inspection. The inspection interval may be set to 1/2 of this dysfunction life, which must be physically demonstrated in full-scale engine tests. Importantly, this approach allows advantage to be taken of new inspection techniques, because inspection capability is one of the major limiting factors of this method. If advantage is to be taken of available safe life beyond 'first crack', the following should be noted:

- In current materials, major cycles are largely responsible for crack-initiation, and minor cycles do relatively more damage in crack-propagation than in crack-initiation. Hence, typically, the mission damage exchange rate ratios may be up to 3 times greater during crack-propagation.
- In the absence of disc removal at the declared life to first crack (LTFC), the number of potentially cracked discs will increase according to a log-normal distribution as lives are extended beyond 'first crack'. This is simply due to the nature of the crack initiation distribution as illustrated schematically in figure 1.
- The approach used is that of sampling without replacement.

Current US practice is to inspect at half of the calculated mean crack-growth life (from an NDI sized defect to dysfunction), and to retire components at a life associated with a probability of cracking of 1 in 1,000. The major exception is for some titanium disks, for which cryogenic spin-pit testing is undertaken to demonstrate that no defects greater than 0.006-in are present in the component. In such cases, the NDI defect size can be replaced by this much smaller size in fracture-mechanics crack-growth lifing-calculations.

# 4. DAMAGE TOLERANCE V LTFC

## 4.1. MATERIALS PROPERTIES

Consider the difference between the Life-To-First Crack (LTFC) and burst distributions for a typical component or test piece. In figure 1 the mean life to a visible first crack is shown as 1.0. The life to burst is about 50% higher, therefore it has a relative mean value of 1.5 times the mean life to first crack. This equates directly to spin-pit testing experience.

For any assumed crack size, the crack propagation life beyond this size would have the form AB, shown in figure 2. However, the line AB specifically identifies the minimum crack size for which, under high cyclic stresses, the whole of the component life is consumed via crack propagation. In practice, this is fully consistent with the 'effective-initial-flaw size' used in the UK fracture mechanics database lifing methodology.

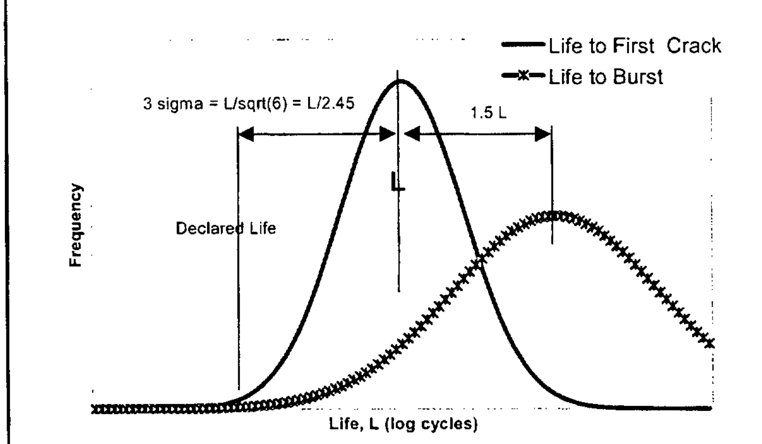


Figure 1 – Idealised population crack and burst distributions

Consider the fatigue curve, which shows the relative significance of the crack initiation and propagation periods. The line A-B is the tangent to the first part of the

failure curve where, typically, the stress cycles are large and failure occurs in less than about 20,000 cycles. $N_p$ represents the number of cycles in crack propagation, and Ni those spent in crack initiation.

Let us now consider the way in which declared lives are

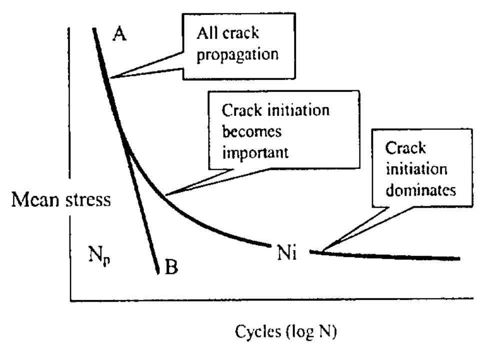



Figure 2 – Life in crack initiation and propagation

calculated. If it is assumed that cracks are detectable at 0.38 mm depth, then from a small sample the declared life can be derived using appropriate statistical procedures as below.

In damage tolerance, with this starter-crack size and using mean crack-growth data, the crack propagation life is calculated. (In contrast to the minimum crack size that *can* be detected by NDE, the damage tolerance approach uses as its starting point the minimum crack size that will *not* be missed.). Typically, 50% of this calculated value is allowed as a service release life. If it is accepted that the difference or scatter in crack growth rates, between the strongest and weakest component, is typically a factor of four then the following applies:

- A component that has a crack size at the NDE limit, that is not detected by inspection, has about a 1 in 750 chance of bursting in service use.
- The mean risk of the population is approximately 1 in 7500.

### 4.2. SUMMARY

Based on the figures illustrated, the damage tolerance release life in cycles can be about 25% of the typical life-to-first-crack (LTFC) release life. However, these comments are based on release life in 'cycles'. For the LTFC case, when the minor cycles incurred, during a service mission, are converted to equivalent major (reference) cycles, the exchange rates typically range from 2 to 5. This is based on analyses of mission usage tapes and is dependent on the severity of the mission flown. In contrast, for the same mission profiles, the damage-tolerance exchange rates may typically be 3 times more severe. Hence, when compared in terms of the actual missions that can be flown, the damage tolerance service release life of 25% of the LTFC release life in reference cycles converts into less than 10% of the LTFC service missions. This corroborates and quantifies the evidence of figure 2. It should therefore be noted that the introduction of damage tolerance is a major step forward in terms of airworthiness and aircraft safety, but this is only achieved through increased 'cost of ownership'.

## 5. CUMULATIVE DAMAGE ASSESSMENT

The main reason for usage monitoring is to ensure that engine components do not fail in service. In addition to the direct airworthiness issue, the consequences of failure can be measured in several other ways including loss of human life, loss of missions, and increased costs associated with unplanned maintenance.

The main problems facing the design team centre on lack of knowledge, about how the engine will really be used in service. Experience shows that most military aircraft undergo changes in the way that they are operated, frequently without the conscious realisation of the operator. The engineer tasked with estimating the service life of each aero-engine component requires information about the intended duty of the engine, the component material, component testing results, field experience, and component management and retirement policy. Figure 3 illustrates the stages involved in such life usage assessments. With full knowledge, the designer can do an accurate job. Unfortunately, the design process is invariably in an open loop situation. (Accelerated simulated mission endurance testing (ASMET), is an attempt to close the loop, at least for 'hot parts'. However, the difficulties of completely predicting service conditions, and testing for them are such that risk cannot be completely removed.)

Initially, the thrust to improve life usage predictions came as a response to the management and logistic needs of in-service fleets. Simple monitors were fitted to engines and data recorded and analysed by the manufacturer. For the first time actual service usage data were available to the operators and the designers to enable realistic life limitations to be imposed. Engine operators count engine-life consumption in (engine flying) hours or (engine flight) cycles, where an engine flying hour just means an hour of engine flight, and an engine flight cycle means simply one flight. In reality, the life consumed during an engine run or flight is based on the effects of the continuously varying stresses, strains and temperatures experienced at critical areas in the components.

The values of these parameters obviously depend on the actual mission profiles, engine intake conditions, individual pilot reactions and several other parameters and are used to assess the low-cycle fatigue, thermal fatigue, or creep service capability of individual components.

The basic steps in cumulative damage assessment are:

- Mission Specification;
- Materials Characterisation;
- Stress and Temperature Modelling;
- Component Testing;
- Field Experience.
- Component Management and Retirement

Damage algorithms are developed in a form capable of application in real time life usage monitoring systems to calculate the consumed life directly from measured engine signals. The monitoring system must be able to respond to the rapid transitions of the input signals as

they appear during rapid aircraft manoeuvring.

Using the measured time histories of engine operating parameters (such as spool speeds, intake and gas path temperatures and pressures) the algorithms calculate the thermal and mechanical boundary conditions for the engine components. These boundary conditions are then used to calculate the transient temperature development within the parts and subsequently the stresses or strains at critical areas. These stress-temperature histories are then used to predict the life consumption associated with each particular flight measured in engine reference cycle damage-related units. Both on-board and ground based processing systems have now been developed. In the former, measurements are processed continuously as they are developed, and in the latter immediately after the end of a flight. The imposed damage is summed over all engine runs to build up complete life consumption records for all monitored parts of an engine. Damage processes which are monitored include low cycle fatigue, thermal fatigue and creep, as appropriate.

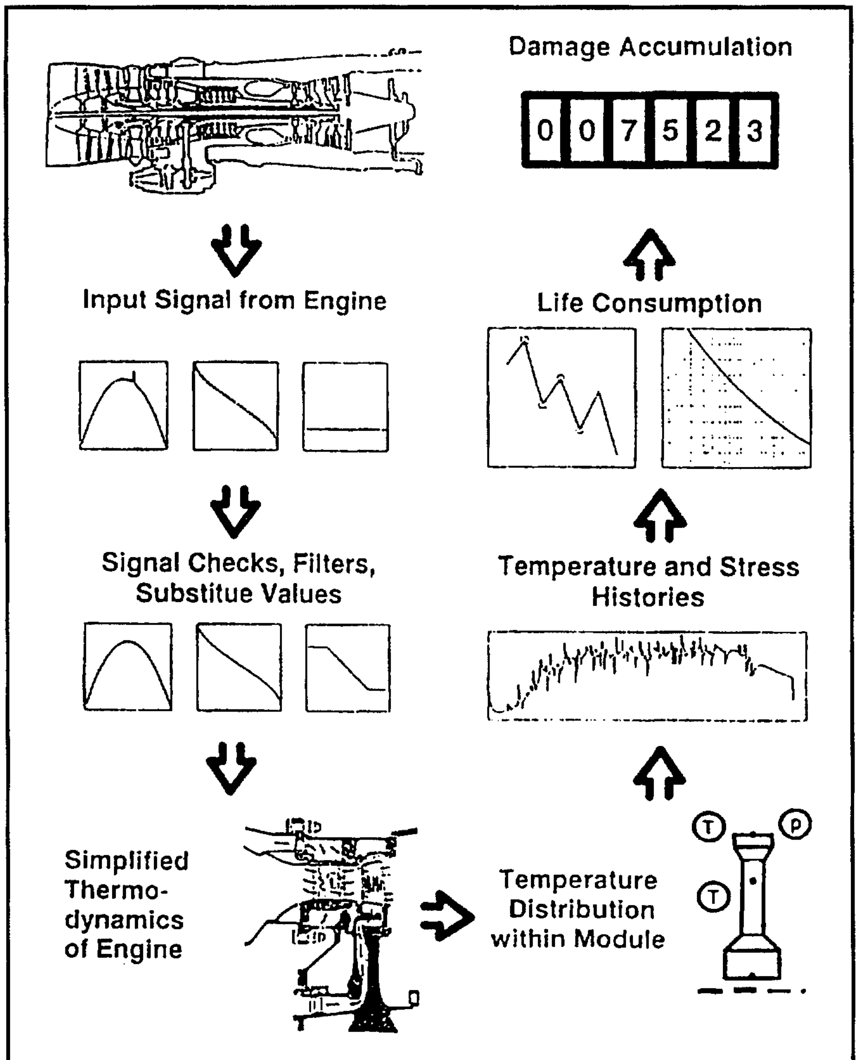



Figure 3 – Overview of Life Usage Monitoring

## 5.1. CALCULATION PROCESSES

The life consumption calculation and monitoring process is determined wholly from engine running history. Generally, this is divided into three parts. With system power up, both the system and algorithms are initialised. The main monitoring tasks immediately follow. Computations are repeated every time step where a typical time step-length may be 0.5 seconds. In the final part (at engine shut down) all the monitoring tasks are finalised and the results obtained are stored. Performance calculation consists of the determination of the temperatures and pressures in the gas path and in the cooling air paths. Calculations also include the computation of torque, bending moments and normal forces in shafts. These form the boundary conditions for the following steps.

### 5.1.1. METAL TEMPERATURE CALCULATION

The temperature distribution of a component is established by sampling at a number of points. For calculation of the metal temperature of a given point the following physical effects should be taken into account:

- Heat transfer to the temperature point from surrounding gas and cooling air and vice versa
- Heat conduction between any temperature differences in the component.

Heat balancing for each of these points may be described by regular differential equations. The coefficients used in the equations for the temperature model can be optimised in such a way that critical basic data used in the lifing calculations are modelled as accurately as possible. In practice, typical deviations from the full mainframe engine simulations are less than 10 K. The coefficients of the equations depend on current engine operating parameters (e.g. spool speed) and the time step-length.

The transient metal-temperature distribution of the component is calculated throughout each of the three phases of an engine 'cycle' (start, run and engine shut down). In phase 1, the initial temperatures for all relevant points are determined. These values depend on the temperature distribution at the end of the last engine run and the period for which the engine has been stopped. In phase 2, the temperatures for all points are updated at each time step. They are calculated using the temperature distribution at the beginning of the specified time step and the thermal boundary conditions operating at the specific time. In phase 3, the temperatures corresponding to shut down stress peaks are computed. Such metal temperature distributions are required for both the calculation of

thermal stresses, and for the temperature influence in damage assessment.

### 5.1.2. STRESS CALCULATION

Stress modelling generally involves finite element stress analyses, determination of physical and thermal boundary conditions, heat flow calculations and the step-by-step assessment of the stresses induced throughout a large number of flights. Transient total stresses are calculated for each monitored critical area by summing up centrifugal stresses, thermal stresses, pressure-induced stresses and any additional stresses. The centrifugal stress is related to the square of the spool speed. Thermal stress is derived from the current temperature distribution. Additional stresses include pressure-induced stress, the effects of bolt clamping, residual stresses etc. If shafts are monitored, stresses due to torque and bending (resulting from gyroscopic effects) may be included as well.

The calculation of total stresses is repeated at every time step during the main phase. In the initial and final phases, the stress peaks which occur during engine start, and after engine shut-down are also calculated. The coefficients used in the stress models are optimised in such a way that given basis data are modelled as accurately as possible. Because in a cyclic damage regime the major cycle contributes most to accumulated damage, the accuracy is steered so that the maximum and minimum stresses are calculated accurately, whereas for medium stress cycles, typically, deviations of 2 % may be accepted. If damage assessment is based on fracture mechanics methods, stress intensity factors or other fracture mechanics based parameters are derived from the stresses.

## 5.2. DAMAGE ASSESSMENT

The established stress-temperature histories are then assessed with respect to the relevant damage mechanisms. In the case of low cycle fatigue damage, stress cycles (i.e. main cycle and sub-cycles) may be extracted using the Rainflow process. The cycles obtained are converted into equivalent damage according Miner's damage hypothesis via materials-specific LCF strain cycles. If fatigue is assessed by fracture mechanics methods, the stress intensity cycles are assessed with respect to the relevant crack growth law (da/dN curve). In both cases, damage due to sub-cycles and main cycles is accumulated over the whole engine run. If creep is an important damage mechanism then in every time step a creep damage increment is also evaluated.

## 5.3. COMPONENT AND ENGINE TESTING

As new component designs are developed, work is often related to the potential advantages offered by new materials. This includes the testing of innovative design techniques to offset any risks associated with the new material. Design validation may be from the qualification test programme. This may involve sub-scale component testing or as with ASMET, seek to prove the structural integrity of the engine hot end parts through full-scale engine tests. Such tests are used to demonstrate that the engine can achieve a full service-life. However, these tests are extremely costly and cannot be used to validate statistically the declared service life. This can only be done through a series of component tests, and statistical assessment of the results. Hence, other basic LCF programmes are used to provide the larger element of proof by analysis, with the model tests providing confirmation and feedback.

In parallel with the engine-component structural-integrity tests, the airborne and ground based monitoring computational system must be qualified. These must be shown to give, within close limits, the same results with the same data as the design models.

### 5.3.1. SAMPLE SIZE

When a test sample is taken from of a population having either a normal or a log-normal failure distribution, its mean and variance are likely to be similar to that of the original population. As the sample size increases, so the likelihood of a similarity increases. The confidence interval (statistical correction factor) associated with any stated confidence level (e.g. 95%) will get progressively smaller as the sample size increases. This leads to the conclusions that:

- The sample must be of adequate size to obtain enough information to show statistically that the sample itself is of a consistent and identifiable distribution.
- The sample can then be assumed to be representative of the original population.

# 6. LIFING ALGORITHMS

The algorithms discussed here relate to the damage induced by the imposed stresses and hence apply in both the mainframe and the Reduced Order Algorithms (ROA) that are implemented in the operational Engine Life Monitoring System (ELMS). Differences between the two sets of calculations relate to:

- Availability of data.
- Measurement accuracy and repeatability.
- Data recording methods.
- Computational speed.
- Computational memory limits.
- Operational requirements and limitations.
- Logistics requirements.

The outcome ranges from simple engine based Total Accumulated Cycles (TAC), to individual component tracking via complex thermo-mechanical algorithms. The derivation of the reduced order algorithms is described in chapter 7.

Historically, such algorithms involve a number of assumptions that include:

- Major stresses are speed related and are due to centrifugal loads, such that the stresses are proportional to the square of the rotational speed, of the component under consideration.
- The major rotating components are most likely to fail in fatigue due to the repetitive loading cycles induced under normal service usage.
- Induced stresses may consist of both steady and alternating components. To ensure that failures do

not occur in service, the allowable alternating stress must be reduced as the steady stress component increases. This can be addressed via a 'Goodman' diagram or similar approach as discussed later. Because of wide variations in observed test results (even with apparently identical samples) the results must be assessed on a statistical basis.

- Fatigue damage is cumulative, in such a way that the fraction of life consumed at any given loading condition can be added to the fraction of life consumed at any other loading condition. This is known as Miner's hypothesis and is discussed later.

More advanced algorithms take into account thermal loads caused by temperature gradients across components, creep and stress relaxation. It is essential that the life usage computations should have mirrored service practice to ensure acceptable levels of accuracy and hence safety. Main differences the more advanced usage monitors are that in contrast to the earlier use of theoretical data, sampled data or values calculated directly from the recorded data can now be used. Measured data may include air temperature and pressure, altitude, engine rotor speed, compressor delivery pressure and outlet temperature. Extracted stress and strain cycles and damage summation are all established using procedures similar to those used in the design stage. However, the analysis models used to calculate transient temperatures and stresses require substantial computing power and large storage capacity. Such calculations can only be effectively done on ground based installations. For on-board processing, it is necessary to construct simplified heat transfer and stress analysis routines that provide an acceptable compromise between computational time and accuracy. Hence, these must be correlated against mainframe thermo-mechanical analyses and acceptable degrees of accuracy established. Formal qualification procedures must be used to verify that the results are robust and directly comparable to those obtained from full analysis procedures.

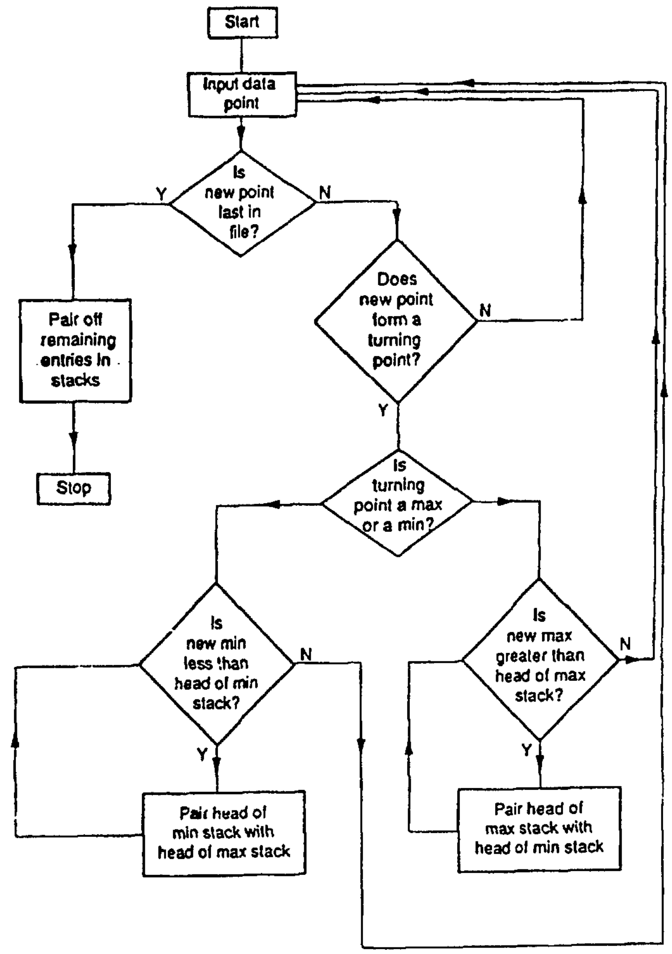


Figure 4 - Rainflow algorithm

## 6.1. MECHANICAL STRESS BASED ALGORITHMS

In the simplest case of mechanical stresses only, these can be related directly to shaft rotational speed. Hence, the results from a stress analysis conducted for a specified shaft speed can be used to determine the critical areas of the disc and the peak stresses operating in these regions. For all other loading conditions, the peak stresses in the disc can be related to the previous identified peak stress or reference stress, $\sigma_r$, by an expression of the form:

$$\sigma_i = \sigma_r \left( \frac{N_i}{N_r} \right)^2 \qquad (1)$$

Where $\sigma_r$ equals peak stress in the disc under imposed engine reference shaft speed conditions $N_r$(rev/sec), and $\sigma_i$ equals stress in the disc corresponding to $N_i$ (rev/sec).

The above expression enables recorded histories of rotational speeds to be used to determine the stress-time profiles experienced at critical locations in the disc throughout any mission profile. In practice, however, the fatigue damage incurred by the disc is not specifically related to these stresses but to the cyclic thermo-mechanical stress-ranges that occur within the stress time history.

It is therefore essential to extract the significant stress cycles from the stress histories of the sortie profiles. For cycle extraction, a procedure such as the Rainflow method is used. This allows the flight profile to be analysed sequentially but requires only very small amounts of temporary data storage and hence is suitable for both ground station and on-board analysis. In practice it has been found that with efficient algorithms, data stores rarely contain more than four or five unpaired high or low turning values before the next set of minor cycles can be extracted. A flow diagram for the Rainflow cycle extraction process is shown in figure 4. The method takes no account of rates of cycling or dwell effects and assumes that interruption of large stress cycles to complete smaller cycles does not affect the damage imposed.

We may wish to assess the effects of a variety of mean stress levels, in the absence of an all-inclusive fatigue database. To do so, an estimate has to be made of the equivalent 'zero-to-max' stress range that would impose an amount of fatigue damage equivalent to that caused by the identified high mean-stress-range test. Probably, the most common empirical approach used to quantify the effects of mean stress level on fatigue is the Goodman diagram.

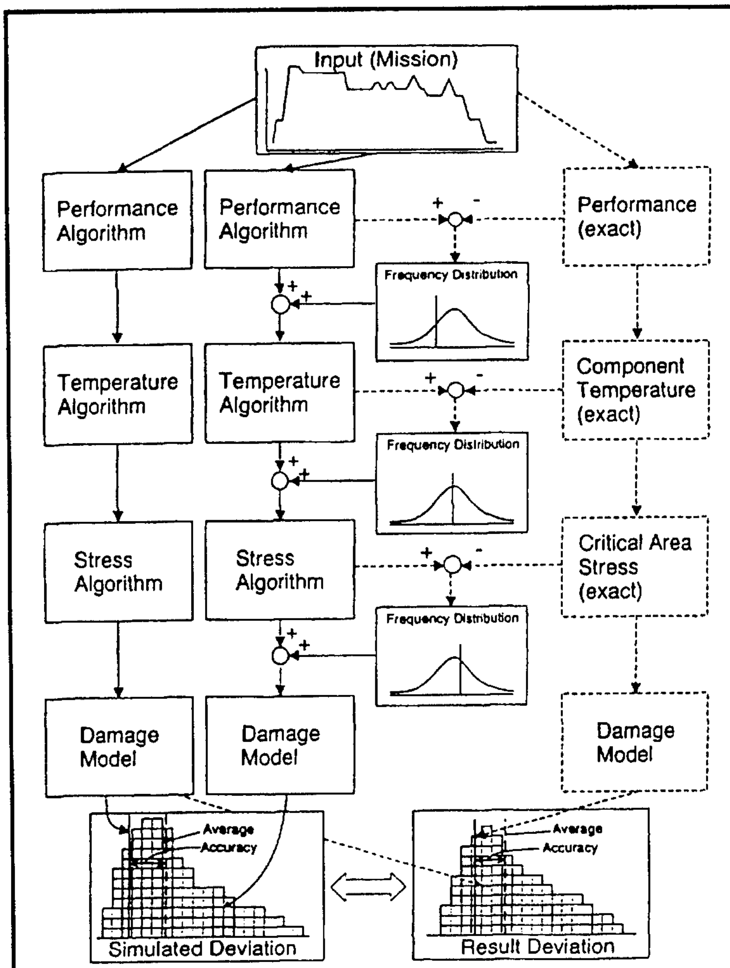


Figure 5 - Overview of the usage algorithms

In this procedure, mean stress is plotted along the horizontal axis and alternating stress range along the vertical axis. A line is then drawn to join the tensile strength of the material to the stress amplitude of the push-pull test having '$N_F$' cycles to failure. It is assumed that all combinations of mean stress and stress amplitude that lie on this line will have a common life to failure. Hence, high r-ratio stress-ranges can be expressed in terms of their Goodman equivalent zero-to-max stress, $\Delta\sigma_E$, via the expression:

$$\Delta\sigma_E = \left[\frac{\sigma_{MAX} - \sigma_{MIN}}{1 - \frac{\sigma_{MIN}}{\sigma_{UTS}}}\right] \qquad (2)$$

Although there has been little experimental verification of this approach, it is widely used in lifing calculations. Equivalent expressions can be constructed for strain based fatigue results and an alternative formulation that accounts for maximum stress and mean strain is the Smith Watson Parameter, $\sqrt{(\sigma_{max}\Delta\varepsilon E)}$

Having identified the equivalent stress range, Miner's hypothesis can be used to convert the damage imposed by such a cycle to its equivalent amount of major cycle damage. The Miner's expression has the form:

$$\frac{\Delta\sigma_R}{\Delta\sigma_E} = \left(\frac{n_{RF}}{n_{EF}}\right)^m \qquad (3)$$

where $n_{RF}$ equals the number of reference cycles to failure at the reference stress, $\Delta\sigma_R$ and $n_{EF}$ equals the number of cycles to failure under the equivalent zero to max minor cycle, $\Delta\sigma_E$. The slope of the fatigue design curve is 'm'.

## 6.2. THERMO-MECHANICAL ALGORITHMS

In practice, the simplified calculation of the engine thermodynamic cycle frequently makes use of compressor and turbine performance maps for gas properties and thermodynamic relationships. From these, the temperature profiles of surrounding areas may be indirectly monitored. When not all components have reached ambient temperature between engine runs, a time-bound estimate of the temperature profile is used. During normal running, temperatures for the identified time-step are derived from previous temperature distributions, and coefficients describing heat transfer and conduction are calculated from the current operating conditions. The algorithms operate in such a way that the steady state temperatures are exact with transients being within about 10°C of their actual values.

The temperature transfer matrix describes heat transfer between hot gas, cooling air and component surfaces, heat conduction and radiation. Generally, in current monitoring algorithms, time-steps, are fixed and temperature changes per time-step can range between about 10°C, after slam acceleration to 0.001°C in the asymptotic phase. Although during a mission, transient temperatures can induce significant thermal stresses, the major transient effects are imposed during the take-off and landing stages in the flight profile. The most sophisticated analysis procedures may involve a full life analysis predicting component transient thermal and mechanical stresses from flight recorded data. The coupled transient temperature and transient stress, finite element analyses require extremely short time steps to ensure adequate identification of both the instantaneous transient thermal stresses and specifically of the turning values associated with the changes in throttle setting.

The procedure is illustrated in figure 5 and involves:

- The computation of gas stream temperatures, pressures and velocities from the known characteristics of the engine;
- The determination of appropriate gas-metal heat

transfer coefficients and transient temperatures throughout the component and throughout the flight;
- The calculation of the thermal stress distribution profiles throughout the flight from these calculated temperatures.

Where transient thermal stresses can make a significant contribution to the overall stress ranges, a large number of computations must be solved rapidly and accurately. This requires significant computing power.

One long established approach to overcome this has been via the use of a simplified thermo-mechanical stress analysis model which is then matched with the full thermo-mechanical finite element computations. Typically, for the centre regions of a large turbine disc, the characteristic response time of the disk can approach 20 minutes. In practice however, in excess of 20 major throttle movements may have occurred during this time. A consequence of this is that most transient events commence from a non-stabilised state. The values of the critical parameters involved can be uniquely established by curve fitting to the results obtained for the detailed, coupled, thermo-mechanical analysis.

The assumption is made that temperatures always decay exponentially towards their asymptotic values (if the defined conditions were held indefinitely). The characteristic time constant of the decay depends on the engine conditions. This can be modelled via a small number of thermal masses (typically 3), with each having 4 or 5 dependable constants to control the temperature at a particular location. Such models can be used to calculate transient temperatures and their associated thermal stresses.

Thus, for illustrative purposes in such an approach, given the spool speed, N, the local gas-temperature, T, and gas pressure, P, for each individual mass: the asymptotic temperatures $Ta_i$ can be related to the gas stream temperatures. This is through a linear expression of the form:

$$T_{ai}(t) = K_i^1 T + K_i^2 \qquad (4)$$

The heat transfer coefficients $H_i(t)$ can be obtained from:

$$H_i(t) = K_i^3 (NP / T^{0.7})^{K_i^4} \qquad (5)$$

and the temperatures of the masses $T_i(t)$ from the numerical solution of the differential equation:

$$\frac{dT_i}{dt} = H_i(T_{ai} - T_i) \qquad (6)$$

with an appropriate initial condition identified for the start of the flight. These equations lead to a thermal stress equation of the form:

$$\sigma_{(t)} = \sum_i K_i^5 T_i(t) \qquad (7)$$

where 'i' is summed over the number of thermal masses that significantly influence the selected location.

In addition to calculating the thermal stresses, it is necessary to retain calculated corresponding temperatures since material mechanical-property data can be highly temperature dependent. It is essential that the model is validated against the stress and temperature results from a full analytical solution. The verification should include features not in the original conditions and must include transients between different engine states, and from non-stabilised conditions. The combined thermo-mechanical stress equation is of the form:

$$\sigma_{equiv} = \sigma_0 + AN^2 + \sum B_i T_i \qquad (8)$$

where $\sigma_0$ represents any residual assembly stresses and the second and third terms are the mechanical and thermal contributions to the total induced stress.

### 6.2.1. CUMULATIVE DAMAGE

For single minor cycles, damage can be expressed in terms of equivalent reference cycles via the expression:

$$n_R = \left(\frac{\Delta\sigma_E}{\Delta\sigma_R}\right)^{\frac{-1}{m}} \qquad (9)$$

Hence, for each mission, the exchange rate β, can be expressed:

$$\beta = \sum n_R = \sum_1^{n_c} \left(\frac{\Delta\sigma_E}{\Delta\sigma_R}\right)^{\frac{-1}{m}} \qquad (10)$$

where $n_c$ equals the number of cycles extracted via the Rainflow analysis of the mission profile, and, $\Delta n_R$ equals the equivalent number of reference cycles that have been consumed. Figures 6a and 6b provide a schematic illustration of the calculation involved in usage monitoring. The references to EUMS refer to the UK Engine Usage Monitoring System, a tool used over many years on many engine types to provide data for all types of engine monitoring work.

When using the basic low cycle fatigue algorithm as described above it is common practice to apply the concept of a 'cut-off stress range' below which the LCF fatigue damage of the disc is assumed to be infinite. This value is frequently set to a stress level at or close to the endurance limit, and leads to a slightly more complex expression of the form:

$$D = \frac{1}{n_{RF}} \sum_1^{n_s} \sum_1^{n_c} \left[\frac{\Delta\sigma_R - \Delta\sigma_C}{\Delta\sigma_E - \Delta\sigma_C}\right]^{\frac{1}{b}} \qquad (11)$$

where $\Delta\sigma_R$ is the reference stress range, $\Delta\sigma_C$ the asymptotic stress range, $\Delta\sigma_E$ the total equivalent zero-to-max stress range and $n_{RF}$ is the number of reference cycles to failure.

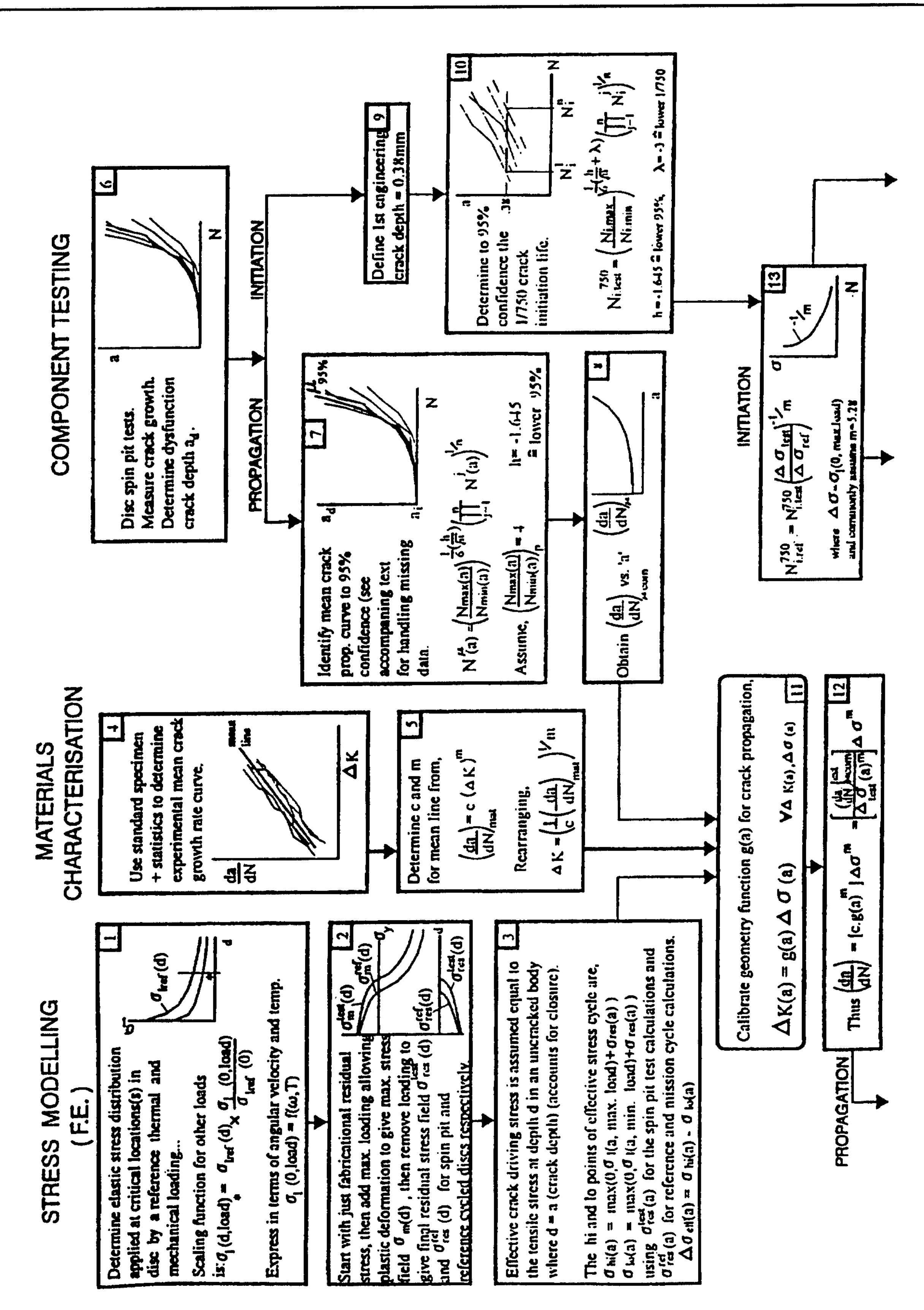



Figure 6a – Overview of usage monitoring procedures

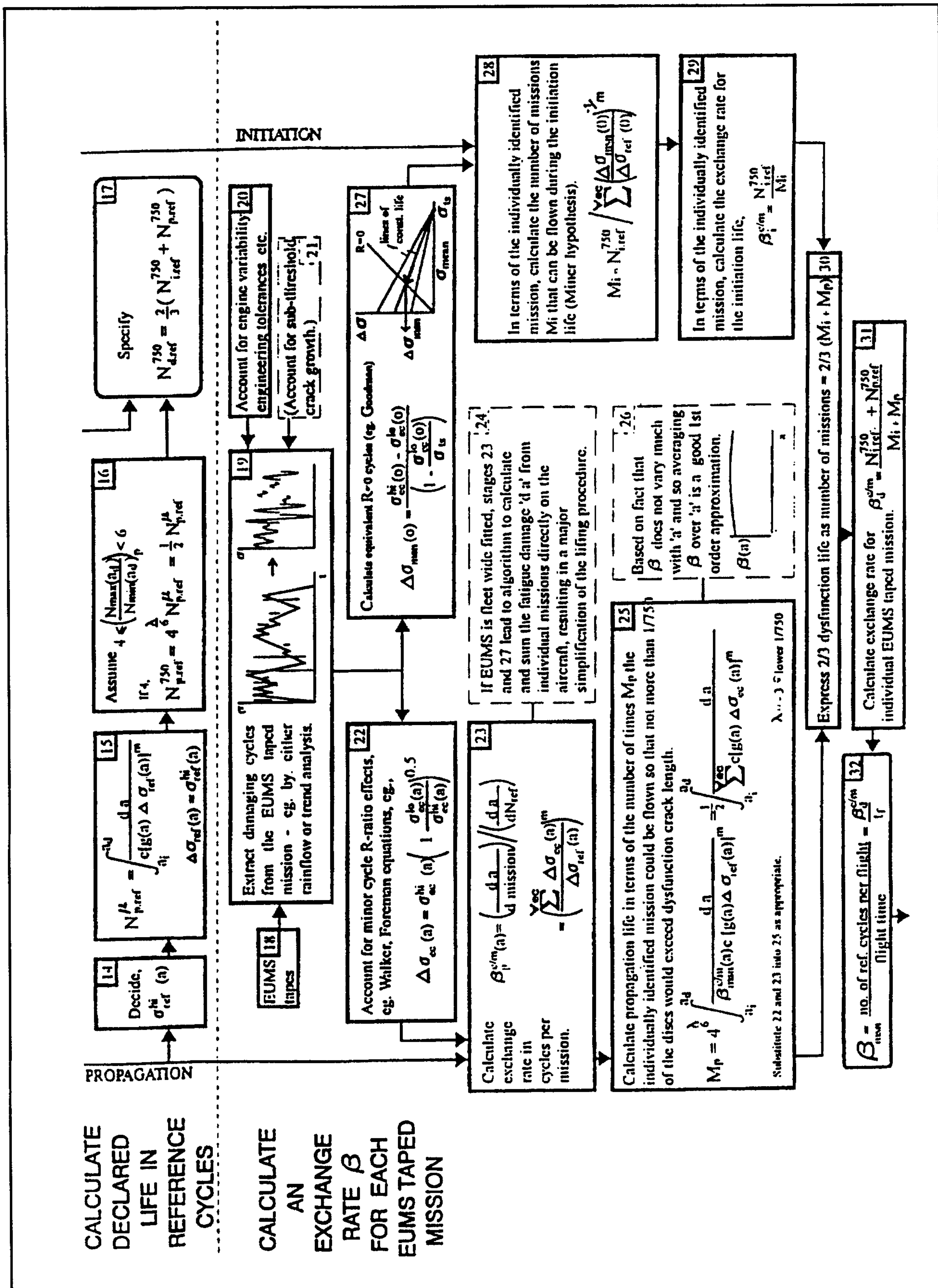



Figure 6b – Overview of usage monitoring procedures

## 6.3. DAMAGE TOLERANCE CRACK PROPAGATION ALGORITHMS

As discussed earlier the basic lifing methodologies are based on a 'Life To First Crack' approach. An alternative, which is increasingly favoured in Europe for new designs is the 'Life to 2/3 Dysfunction' approach. This section describes the algorithms used in this latter approach. They depend heavily on modelling crack growth. In the application of damage tolerance concepts, in contrast to crack-initiation lifing concepts, the size of the crack is now a measure of life consumption. The prediction of possible crack size considers the 'worst case' conditions. Practically, predicted crack sizes significantly exceed those of any identified cracks. Crack size itself is of no practical use to the customer, except as a criterion for component removal. Therefore, based on the correlation between accumulated cycles and crack size, the monitoring system can easily convert the 'crack size' damage into reference cycle damage. This permits a single unit to be used as the measure of life consumption, because within the monitoring system, safe-life and damage-tolerance lifing concepts can be algorithmically combined.

### 6.3.1. FACTORS IN CRACK GROWTH

A large number of scientific papers have been written on fatigue crack-growth models. However, little attention has been given to their application to real components in service. Each factor affecting crack growth must be identified and statistically modelled to ensure that the approach can be implemented without compromising current levels of safety. The sequence of stages involved, together with the associated statistically adjusted damage algorithms are illustrated in figures 6a and 6b. The procedures are equally applicable to life extensions beyond 'first engineering crack' and to the ENSIP damage tolerance concept.

### 6.3.2. IDENTIFICATION OF CRACK GROWTH RATE CONTROLLING PARAMETERS

In practice, for laboratory specimens, crack initiation and growth processes under constant amplitude loading are reasonably well understood and they can generally be quantified via the use of linear-elastic fracture-mechanics concepts. The transition to variable-amplitude loading cases, such as occurs under service conditions necessitates the identification of a scaling parameter. Although, in principle, the stress intensity factor could be used, in practice for multi-axial loading and under both global and crack tip yielding, this is a complex technical step and accurate quantitative predictions is difficult.

The simplest and most conservative approach is to assume that for each cycle the full elastic load range, normal to the crack, contributes to the effective stress intensity cycle providing the driving force for the crack growth increment. However, this can lead to inaccurate estimates of the available safe service life because the higher stresses imposed under laboratory test conditions may cause local yielding, and on unloading to greater crack closure. This may result in more extensive compressive residual stresses than are induced in a service component.

### 6.3.3. ROLE OF RESIDUAL STRESSES IN CRACK GROWTH

There is considerable evidence to suggest that a major cycle that induces significant plastic strain at the tip of a crack, is likely to slow down crack growth during the immediately following minor mission cycles. That is, the position and magnitude of the maximum stress in the missions to be flown is dominant in identifying both the boundary of the plastic zone, and the associated compressive residual-stress field when the major cycle is completed. In such cases the following apply:

- Where the loading conditions are high enough to cause local yielding at a stress concentration in the uncracked body, any subsequent crack growth will only occur when the crack is fully open.
- The stress level at which this occurs depends on the magnitude of the local compressive residual stress field created by a preceding overload. The effective stress-intensity range driving the crack growth process is simply the peak elastic value associated with the maximum stress, minus the crack opening stress level.
- Where local plastic yielding can occur, due account must be taken of the relative effects of the different plastic zone sizes created under spin pit component life determination and under service conditions.
- The lifing methodology for crack growth must therefore be more complex than that for the crack initiation situation.

In any fracture-mechanics based crack growth methodology, it is essential to identify an appropriate statistical model so that current safety levels are not compromised.

## 6.4. FIELD EXPERIENCE

This is the point at which the success or otherwise of the design is evaluated in earnest. Unfortunately, several of the original design assumptions may no longer be completely valid by this stage. Almost invariably, the way in which the engine is actually used will be significantly different from the earlier assumptions and the recipient may want to exploit to the full all of its new or unique capabilities.

At this stage, components may require a modified design usage model. The first indication of this may come from the usage data collected from the engine monitoring system. It may be seen that the rate of damage accumulation is not always that which was expected. As experience builds up, the cumulative damage picture across the fleet will also build up and special cases that cause very high or very low levels of damage may be identified. In some cases, this may lead to changes in pilot practice, or in others to changes in spares provisioning rates.

More importantly, the information that is being gathered provides a model of component usage based on the indirect measurements and computations of the airborne system.

The results indicated by the monitoring system only work

according to the rules built into it. Any hidden flaws or errors in the original model calibration will not reveal themselves unless a component fails. One way around this is to remove a sample of components from service and test them to destruction under controlled conditions. This, however, is expensive.

## 7. SUMMARY

Few tasks in engineering are more critical than the prediction of the service lives of fracture critical components and the complementary task of prediction of component service-life consumption. Predictions of the state of stress and of strain are made difficult by intricate geometrical features and by uncertainties in transient temperature distributions. The effects of localised creep and engine performance deterioration also cause this stress distribution to change significantly with time. In recent years proposed lifing models have become progressively more sophisticated, with more of the identified lifing variables being incorporated.

It is clear that any model aimed at producing acceptable life predictions must examine the loading cycles in detail, and take into account variables such as the stress and inelastic strain at each critical area of the disc. Specifically with respect to fracture critical components, safety cannot be compromised. Hence, to utilise the maximum available lives of such components it is imperative that individual component service-life-consumption is tracked. This chapter has attempted to illustrate the general principles involved in usage monitoring with simple illustrations of their application.

## 8. RECOMMENDATIONS

As more lifing variables are identified and taken into account, it is essential that sufficient material properties data is available to the designers. This may necessitate increasing investment in materials databases, through both the ENSIP and the European database approaches described above.

Normal variations in LTFC initially make a damage tolerance approach to design appear attractive. However, some advantages may accrue from selecting the approach on a case-by-case basis.

## 9. REFERENCES

G. F. HARRISON, M. E. F. SMITH, J. NURSE: Procedures for Aeroengine Component Life Usage Prediction, AIDAA/AAAF/DGLR/RAeS 5th European Propulsion Forum, Pisa, Italy 5-7 April 1995.

G F. HARRISON, P.H. TRANTER, S.J. WILLIAMS: Modelling of Thermomechanical Fatigue in Aero Engine Turbine Blades.

G. F. HARRISON, A. D. BOYD-LEE: A Damage Tolerant Lifing Methodology for High Strength Disk Alloys. DRA/SMC/CR942045/1, October 1994.

In turn the following references are made:

G. F. HARRISON, J. E. GOODWIN, R. S. DENNIS: Laboratory simulation of thermal fatigue, Int Conf Society of Environmental Engineers Fatigue Group V1 No 16 1-14 1976

G. F. HARRISON: Deformation and fracture of some nickel based gas turbine alloys, PhD thesis, University of London 1975.

G. F. HARRISON, M. G. COCKROFT: The effects of cycle parameters on high temperature low cycle fatigue, 1974, AGARD CP 155, Paper 4 1-13.

F. HARRISON, G. P. TILLY: The state and cyclic creep properties of three forms of a cast Nickel Alloy, Int Conf on Creep and Fatigue in elevated temperature applications. 222.1-9. I Mech Eng, Sheffield, 1974.

F. COFFIN: Fatigue at high temperature, ASTM STP 520, 1972, 5-34.

S. S. MANSON, G. R. HALFORD, M. H. HIRSCHBERG: Creep fatigue analysis by strain-range partitioning, NASA TM X-67838, 1971.

G. R. HALFORD, J. F. SALLSMAN, M. H. HIRSCHBERG: Ductility normalised strain range partitioning life relationships for creep-fatigue life predictions, NASA TM 73737, 1977.

T. KITAMURA, G. R. HALFORD: A non linear high temperature fracture mechanics basis for strain range partitioning, NASA TM 4133, 1989.

R. S. NELSON and J. F. SCHOENDORF: Creep fatigue life prediction for engine hot section materials (isotropic) - fourth year progress review, NASA Report N89-12914, 1986.

J. BRESSERS and M. M. BUCKLEY: Thermomechanical and isothermal fatigue of coated and uncoated nickel-based single crystal superalloys, Rolls-Royce plc, private report, 1990.

G.F. HARRISON & D P SHEPHERD: Lifing Philosophies for Aero Engine Critical Parts. Aero Engine Reliability Integrity and Safety. Proceedings of the Royal Aeronautical Society, 1991.

P. WATSON & B.J. DABELL: Cycle Counting and Fatigue Damage. Jnl. of Soc. of Environmental Engineers, Sept. 1976.

T.M. EDMUNDS & R.A. LAWRENCE: Monitoring Engine Thermal Stresses. AGARD CP386 Engine Cycle Durability by Analysis and Testing, Netherlands 1984.

H. PFOERTNER & C. ROSS: Preparing Life Usage Monitoring for the next Decade. Proceedings of the 18th Symposium. Aircraft Integrated Monitoring Systems. 19-21 September 1995. University of Stuttgart.

J.BROEDE & H. PFOERTNER: Advanced Algorithm Design and Implementation in On-board Microprocessor Systems for Engine Life Usage Monitoring. Proceedings of the 15th Symposium. Aircraft Integrated Monitoring Systems. 12-14 September 1989. University of Aachen.

J.BROEDE , H. PFOERTNER, K. RICHTER The Importance of Testing for Successful Life Usage Monitoring Systems. Proceedings of the 19th Symposium. Aircraft Integrated Monitoring Systems. 4 - 7 May 1998. Garmisch - Partenkirchen, Germany

# Chapter 7
## Lifing Procedures and Monitoring System Verification and Validation

by

*(P.-E. Mosser and P. Everitt)*

# 1. ENGINE LIFE MANAGEMENT

## 1.1. INTRODUCTION

The life control process for critical parts begins with the definition of an appropriate safe or economic life for a particular component. The design is then done using available engine models and material property databases. The models and databases now available in industry are extremely comprehensive, and almost unrecognisable from those of the early days of the gas turbine.

Checking and testing the various models against the real world for accuracy and repeatability is known variously as verification and validation. These words refer to the veracity or truth of the model or simulated process. Unfortunately, use of the words verification and validation varies, and we need to define how they are used in this paper and elsewhere in this document. For our purposes:

- Verification is checking the truth of an element of a system.
- Validation is checking the truth of a complete system.

These definitions are as used in ISO 9000, which defines Quality Management Systems.

After manufacture, the life consumption of the in-service component is then monitored against the design safe-life using one of the approaches described in this document. Most of the approaches have been used extensively for many years and involve largely 'manual' processes, which do not pose any new validation or verification questions. The most recent approaches use a set of simplified models, known as reduced order algorithms. These are implemented in a suitable recording and analysis system to calculate the life consumed for each individual component. This technique presents significant validation and verification issues that must be solved before the system can be used for the executive control of critical component lives. This chapter concentrates on the validation and verification processes for these systems.

A complete usage monitoring system contains the following functions:

- *Data Recording* - acquisition of the necessary data for calculation of the component usage values.
- *Damage Calculation* - calculating the life used by each component or critical area.
- *Life Management* - Organising the life usage information to support decisions on aircraft deployment, component retirement, engine removals and engine and spares management.

The relationship between these three functions and component life is shown in figure 1.

The damage calculation processing may be carried out on-board or on the ground. In either case, there will be both airborne and ground based elements of the system. Figure 2 shows these arrangements, and figure 3 shows some possible data transfer methods.

The basic calculation methods and processes for predicting the life of a component and for establishing an appropriate 'safe life are covered in chapter 6. In this chapter, we consider the safe-life limit set through the methods of chapter 6 to be totally valid, and describe the derivation and verification of a set of simplified models that are used in-service to estimate life consumption. These models are referred to as the Reduced Order Algorithms (ROA). They are used to perform the 'consumed life' calculations in a practical size of processor. We also describe the validation process that is necessary to demonstrate that the total 'in-service' system provides a satisfactory representation of the full design model and process.

## 1.2. ESTABLISHING THE AUTHORISED LIFE OF LIMITED LIFE COMPONENTS.

Where the consequences of a component failing in a life-related mode are critical, an in-service life limit is imposed to control the risk of failure occurring. This 'authorised' life limit is imposed by the design or airworthiness authority, depending on circumstance, and is set at a lower level than the predicted safe-life because of the need to provide a safety margin. This margin is designed to account for:

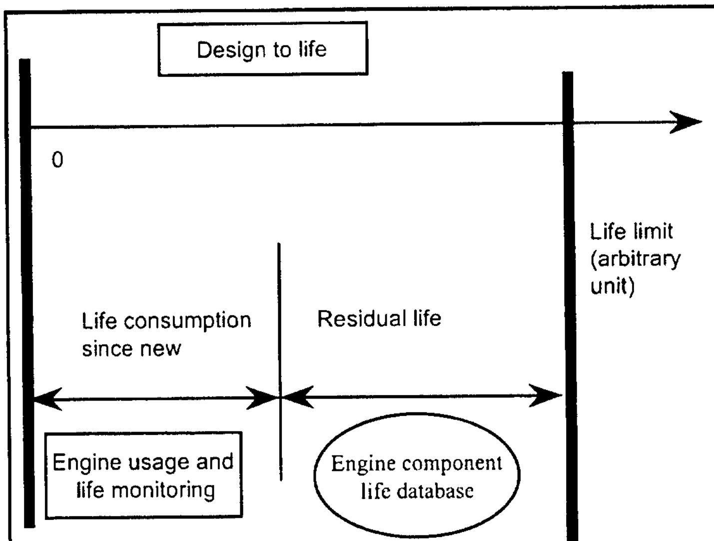



Figure 1 – Designing for life and monitoring usage

- Uncertainty in the design methods.
- Scatter in part dimensions and material properties.
- Ageing and damaging effects from the environment.
- Variability in both engine response and usage.

The evaluation of the margin between predicted life and authorised life is part of the lifing process. The method of evaluation can vary:

- Depending on the particular experience of the manufacturer.
- Depending on the design year of the engine; The life of a recently developed engine type will rely on a certain amount of sophisticated mechanical analysis plus some component and engine testing. Older engine types will be assessed through a smaller number of elementary analyses and more extensive

continuous testing of whole engines.

Nevertheless, certification authorities for both military and civil engines must approve all methods.

## 1.3. REDUCING THE MARGIN BETWEEN ESTIMATED AND ACTUAL LIFE CONSUMED

Improving the design methodology and manufacturing processes can reduce the life uncertainty due to the design methods and to the scatter in part dimensions and material properties. A good design may reduce the impact of ageing and of the environment on the engine. The flight profiles used in life design can be chosen so that they describe more precisely the typical service flights. However, as described in chapter 9, the variability of engine usage and response induces a very large scatter in the conversion from number of flights or flight hours to accumulated damage.

The most satisfactory way to reduce the uncertainty in estimating the effective sorties performed by an engine is to continuously monitor and record enough data to compute and record the damaging effect of each successive flight.

## 1.4. MAIN SPECIFICATIONS FOR AN ENGINE LIFE MONITORING SYSTEM

Let us first establish the differences between the lifing procedures and methods and the Engine Life Monitoring System (ELMS). The correlation between the lifing and the monitoring computation procedures is represented in figure 4. The first column of processes represents the 'Perfect' model and the second the reduced order algorithms.

### 1.4.1. LIFING METHODS OBJECTIVES

The aim of the lifing procedures in the engineering department at any engine manufacturer is to detect all critical areas of all the components of an engine and to compute their life. To do so, they use finite element models of engine components or assemblies to perform transient thermo-mechanical analyses for simulated flight profiles, which cover the expected flight regimes of the engine. The damage analysis procedures then predict for each stressed location the critical damaging mechanism and the corresponding life. This analytical prediction is demonstrated through testing of individual components or whole engines as described in chapter 6. The authorised life is declared at the end of the lifing process by the design or airworthiness authority, depending on national practice.

### 1.4.2. ELMS SPECIFICATIONS

The main objective of an ELMS is to compute the residual safe life of a part or a module after every flight. The design models are too complex and too large to be run in real time even using a large mainframe computer. Therefore, there is a need for simplified models that can operate in an "on-board' computer or in a typical PC at an operating squadron.

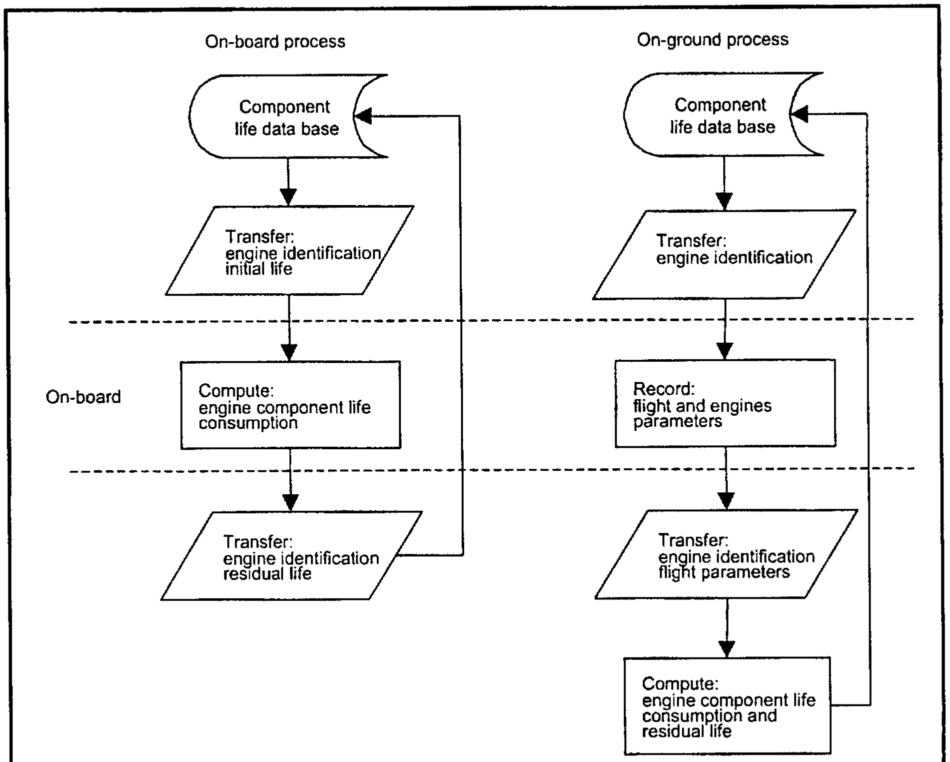



Figure 2 – Alternative architectures for data processing and transfer

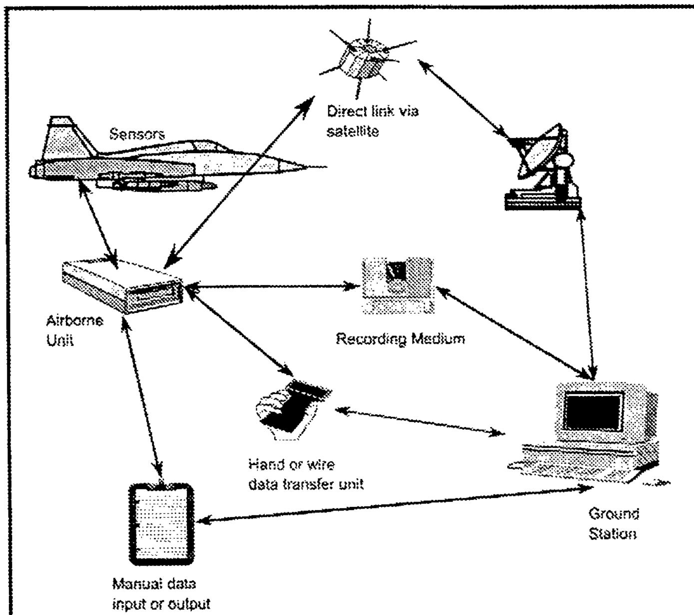



Figure 3 – Data transfer methods

Usually, such a system can be split into two main parts:

- The computation of the consumed life for each monitored engine or component.
- The calculation, storage and management of residual life for engines and components.

The two sub-systems must be able to exchange information. After the building or rebuilding of an engine the reference identity, and life consumed for all the monitored components of the engine should be sent to the damage computation unit. At the end of each flight or suitable number of flights, the consumed life for each engine component is transmitted to the life database. The maximum period before downloading the flight or damage information to the ground station is important regardless of where the data processing is carried out. This is because it affects the need for memory and the practicality of operating the ELMS in service. It is critically important during the design phase that the computational power and memory needed for processing and storage is accurately determined, wherever the various functions are to be eventually implemented.

The reduced models used for life computation should be simple enough to be performed either on-board or at the Squadron unit at the end of a flight or recording period. They should incorporate the maximum possible physical basis to make the demonstration of their validity easier. They should also be consistent with the engineering lifing method used by the manufacturer of the engine.

There is a range of complexity which can be incorporated into these reduced order models depending on the availability of the input data and, particularly for older engines, on the availability of a full finite element component analysis. A relatively modern engine would have sophisticated models already available. In this case the reduced order model should incorporate this and should therefore provide an accurate measure of component usage. Some older engines (50's and 60's) may not have been through a re-evaluation process that will generate modern stress models, and in these cases, it may be more appropriate to use simple algorithms based on the existing stress models. Appropriate margins will have to be incorporated into these very simple algorithms, but they will still give significant benefits compared to the most basic life control methods based on flying hours.

### 1.4.3. EXPECTED BENEFIT OF AN ELMS

A number of benefits can be derived from a good life monitoring system:

- Life monitoring results in a better knowledge of effective flight parameters and their impact on component damage.
- Overhaul of critical components can be undertaken closer to the life limit, with the same degree of confidence in the safety of the process.
- The ELMS provides practical help for predicting the time remaining until overhaul is required and through trend analyses it allows a precise usage rate to be computed for each engine.
- Knowledge of individual engine life-consumption can be used for fleet-wide scheduling of inspections, overhauls, and aircraft deployment to remote sites.

Due to the scatter of the behaviour between any two engines, these benefits can only be maximised through the fleet-wide use of monitoring units, and ground teams trained to use such systems. Chapter 9 gives an indication of the degree of benefit to be gained from various levels of monitoring.

## 2. DEFINITION OF DAMAGE UNITS AND LIFE UNITS

### 2.1. SQUADRON BASED DEFINITION

There are two purposes to which usage monitoring is put in service operation:

- The engineering officer at the base or squadron needs to know how much residual life is left so that he can decide on the operational capabilities of each aircraft, e.g. can it complete a proposed detachment.
- The logistics specialist at headquarters needs to know how much of the authorised life has been used at any time so that he can order the necessary spare parts. He also needs to schedule engines into the repair shop for replacement of components whose life has been used up.

A variety of different units and counters are in use, sometimes simultaneously, to describe engine usage. Depending on the aircraft operator one can find the following counter types:

- Clock time, total running time, after-burner running time, running time over X% of nominal, etc;
- Number of flights, number of take-offs etc;
- Number of engine-starts, or after-burner light-ups.

There are difficulties in translating such counter information into residual life. This has always applied to military aircraft operations because of their inherent variability, due to 'non-standard' routes and the need to respond to the unexpected. These difficulties increasingly apply to civil operations because they, also, are becoming more variable with the increasing use of flexible ratings.

### 2.2. MECHANICAL ENGINEERING BASED DEFINITION

The damage sustained by a part is not a uniquely defined material property. In practice, it is defined through several different mechanisms such as low cycle fatigue, thermo-mechanical fatigue, creep, and oxidation together with a combination rule as described in chapter 5. According to these models, a part will break when after a history of being stressed at different levels and different temperatures the value of the accumulated damage variable reaches a predefined limit. By observing trends over time this information can provide a good indication of the right time to buy new parts.

### 2.3. PRACTICAL CHOICE

Practical consideration leads to the use of different units for residual life evaluation and for the expected time before overhaul. The principal considerations are given

below:

- However sophisticated the life management method may become, it is necessary to keep some existing global simple usage measures because they are the only systems to rely upon if the high technology is unserviceable.
- Normally the *life limits* of an engine component are expressed in terms of reference cycles defined by the manufacturer.
- In the engine life monitoring system, the *accumulated damage* is computed in the appropriate engineering units, such as LCF cycles, turbine creep units or other continuous damage variable. For homogeneity reasons, it will be translated into the same reference cycles as have been used for declaring the authorised life.
- The residual life is determined in flying hours or number of sorties, depending of the aircraft operator, using "temporary" exchange rates. The Engine Life Monitoring System installed in each particular squadron allows a local "temporary" exchange rate, that can periodically, or continuously, be reassessed instead of the "normal" exchange rate based on overall experience.

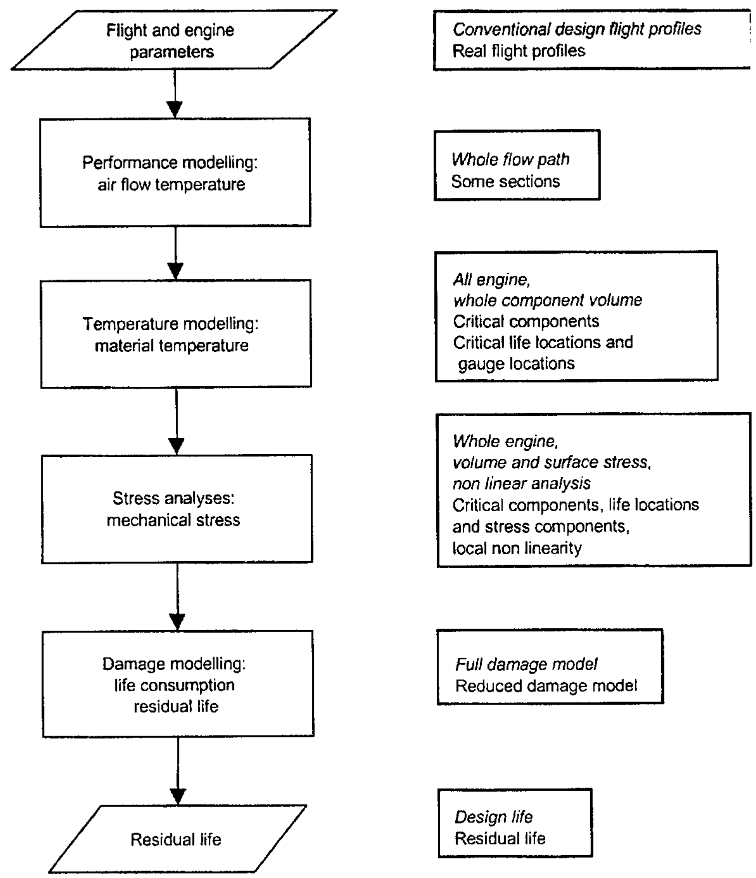


Figure 4 – Comparison of the reduced model with the life design process

## 3. REDUCED DAMAGE MODELLING FOR ELMS

The development of a reduced damage model is usually a two-step process:

- Development of the algorithms. This is usually performed using a conventional computing system such as mainframe, workstations or PC's. This system will be kept and maintained so that it can be used later for modification or development of the reduced models and ground based life studies during engine service life.
- Implementation of the models into operational hardware.

This section will describe development of the algorithms. Implementation will be addressed later.

### 3.1. REDUCED ENGINE DESCRIPTION

Models should be defined in conjunction with the decisions that are intended to be made as consequences of the outputs. Here the model monitors the life consumption of the critical parts of the engine rotor. The decision will be when to inspect or to overhaul the rotor.

Four basic requirements should be included in the monitoring system and the reduced engine model. These are:

- The consumed life should be available after each flight. The damage computation may be made available in real time using an on-board computer or very rapidly after the aircraft lands using a normal PC type computer.
- When transferring damage or flight data from the engine to the ground the monitored components should be unambiguously identified and associated with the data. Engine configuration should be included within the database of the reduced engine description.
- Frequently, only one critical feature may exist in each module. If so, only this critical area need be monitored. For hot sections, there may be more than one critical feature, depending on whether the major damaging mechanism is creep or fatigue. In this case, enough features must be monitored to ensure the integrity of the modelling process. Figure 5 displays a typical representation of the monitored areas included in the ELMS.
- The ELM model must take into account the engine operating conditions, and the mechanical properties of the materials and components. The analytical process to compute the life consumption from the engine parameters is represented in figure 6.

Points 1 and 2 are mostly related to the hardware organisation and the monitoring process and will be referred to in the second half of this chapter. Validation of points three and four will be the main topics of the following sections.

## 3.2. REDUCED ORDER ALGORITHM INITIAL VALIDATION

### 3.2.1. OLDER ENGINES

The reduced modelling appropriate for an engine will depend on the design methods used to predict its component lives. In practice, it will depend on the date of development of the engine. For the older engines designed using analytical computations calibrated by testing, the ROA can only be an application of the rules that used engine-operating parameters. Usually, the ROA will then be a direct computation of consumed life from the maximum rotor velocity during each flight.

For this single step approach, the validation procedure is to run the algorithms through a standard sortie that covers all of the expected flight conditions and exercises all of the logic in the algorithm. All of the flight conditions in the sortie are computed, both by the full design models and by the reduced order algorithms. The results at each flight point are compared and assessed against the accuracy requirements. This gives a large sample covering a wide range of conditions and effectively covers the validation activity for the reduced order algorithms.

### 3.2.2. NEWER ENGINES

The newer engines are hotter and more highly stressed than the older ones, and they are expected to achieve higher levels of safety. They are designed using comprehensive models that start with the instantaneous operating state and predict successively the air temperature within the engine, then the material temperature, then the mechanical stress and finally the increase in accumulated damage. Chapter six describes this process in more detail. The reduced order algorithm will be divided into modules that generally copy the structure of the life design model of figure 4. Each module will incorporate reduced models, based to some extent on the same physical realities and approaches as the full models. The extent of this depends partly on the philosophy of the engine manufacturer concerned. Inevitably, some parameters in the reduced algorithms will require optimisation during the algorithm development process.

A brief description of the essentials for the reduced model, including some important points for their calibration, will be given in the following sections.

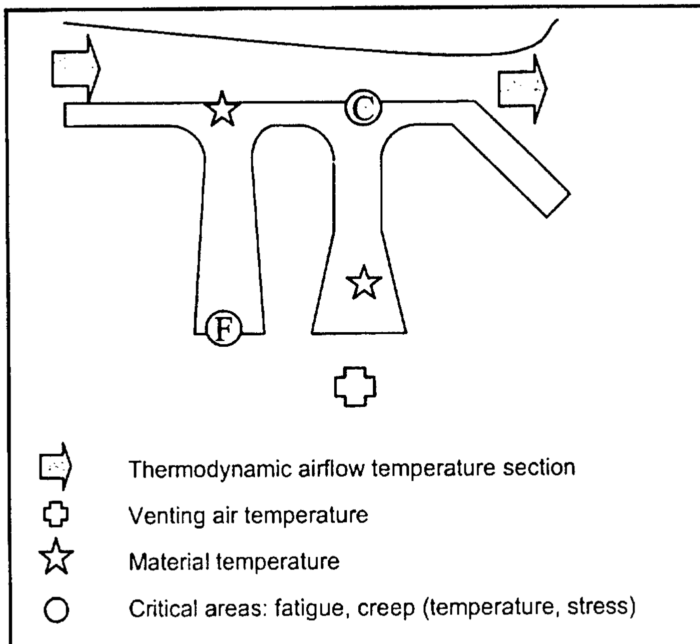



Figure 5 – Reduced module model

#### 3.2.2.1. *MATERIAL DAMAGE MODELLING*

The active damaging mechanisms are determined during the life design process for each critical location. The damage mechanism is usually fatigue or creep, and some new algorithms incorporate crack propagation. Allowances for crack initiation are more difficult. They depend on the lifing philosophy, and the experience of the manufacturer, and are unlikely to be included in a reduced model.

Figure 6 shows the generic organisation of such a model.

There are inherent differences between the design model and the reduced model. Firstly, the material database incorporated into the reduced model is a simplified derivation of the design material database. This is to reduce the computer memory required by the ELMS. Secondly, the thermo-mechanical history of a real engine is much more erratic than the conventional flight profiles used in design. The life modelling and the material database should accommodate:

- Numerous small cycles of real flight;
- Very rapid loading which occurs during rotor acceleration and deceleration;
- Constant loading can lead to some concerns in real time creep and fatigue modelling.

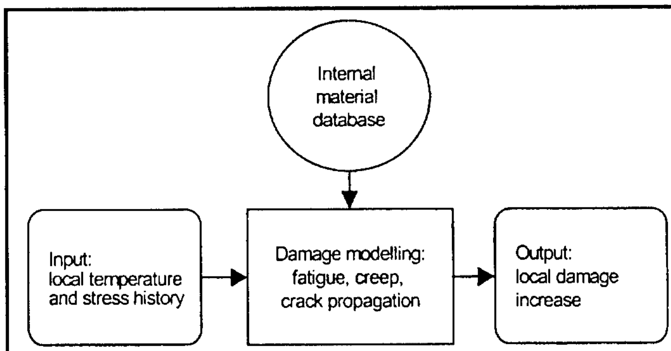



Figure 6 - Materials Damage Modelling

Therefore, to be sure of the equivalence between the design and the reduced models, all the verification procedures of the reduced damage modelling are to be conducted for each material on a wide range of flight profiles. This will include real flights, and use the same sampling frequency as will be used in the life-monitoring device.

#### 3.2.2.2. *STRUCTURAL MECHANICS*

Centrifugal forces and temperature gradients cause the main loads on a jet-engine rotor. For the discs and shafts the simplest equation, which can describe the generated stress at one location is of following type:

$$\sigma = \sigma_0 + A \times N^2 + \sum B_i \times T_i$$

Where $\sigma_0$ is the assembly stress, N the rotational velocity and $T_i$ is temperature at specific gauge locations of the rotor. Typical gauge points are shown in figure 7.

The reduced mechanical model is fitted to the design model by adjusting the number of temperature gauge points, their locations and the model coefficients. The aim is to minimise the difference between the stress levels resulting from this model and those from the design model.

Whenever needed this basic equation can be altered to cope with:

- Plastic loading;
- Overload detection;
- Modification of boundary conditions;
- Introduction of pressure loading.

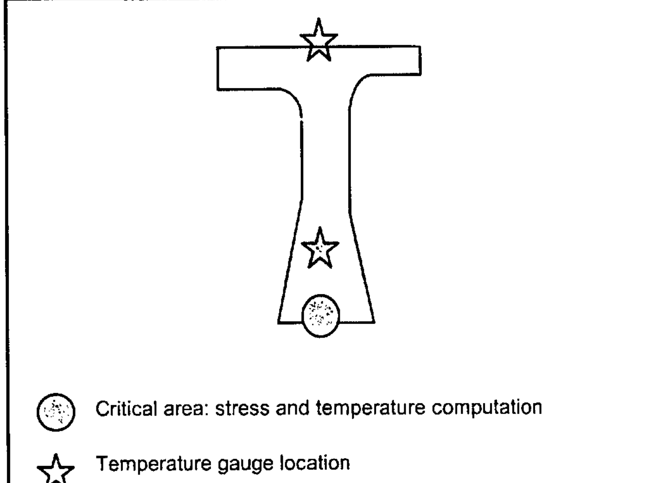



Figure 7 – Reduced thermo-mechanical model of a disk

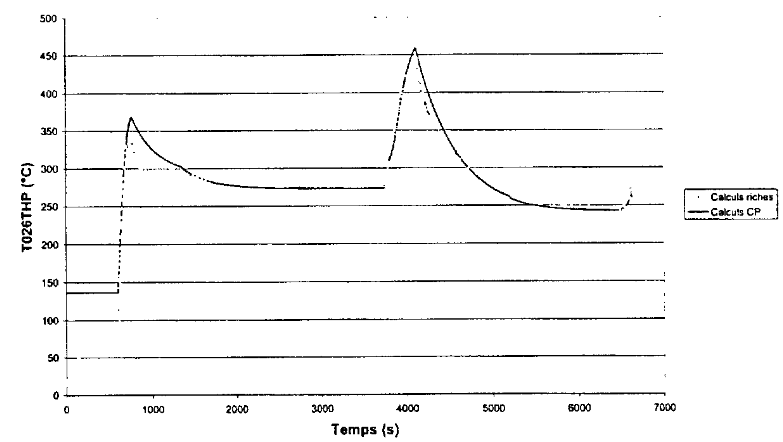



Figure 8 – Reduced model results compared to the full model

A similar equation can be written for torque loading.

The verification of the accuracy of the structural model should be performed on a specified variety of flights to ensure that all the possible temperature situations have been covered. In fact, the higher the stress level and the more frequent the occurrence of the situation leading to this level, the more stringent is the need for a precise reduced model. Despite the apparent simplicity of the equations, it is important to keep in mind that this type of equation is a carefully constructed representation of a very complex model. The temperature distributions and centrifugal loads used during the model fitting must be representative of real situations. If they are not, then the reduced model can be exaggeratedly complex or even inaccurate in the temperature range encountered in service.

### 3.2.2.3. MATERIAL TEMPERATURE MODELLING

The locations where the material temperature characteristics have to be modelled are:

- The critical points where life is calculated;
- The gauge points as needed by the stress calculations.

Most of the time the transient temperature calculation is a 2-step process. At each instant first the steady state temperature is computed, and then the temperature change rate. The inputs are the power setting, the air flow temperatures and the rotor speeds. Figure 8 shows the accuracy it is possible to achieve by this method for a turbine disk location.

Two important factors can affect the accuracy of the reduced thermal modelling:

- A bad choice of model parameters;
- The full model does not become a 'perfect' representation of the engine until almost the end of the development programme.

It should be noted that the correct prediction of thermal stresses requires the accurate prediction of both low and high temperature conditions.

### 3.2.2.4. THERMODYNAMIC MODELLING

The aim of this model is to provide the air temperature characteristics needed by the solid-component temperature model. Again, it is important to note that the correct knowledge of thermodynamic state of an engine will be available only after having investigated the full flight envelope.

### 3.2.2.5. CONNECTING THE REDUCED MODEL ELEMENTS

Some decisions concerning the structure of the whole model should be taken before finally assembling the individual sub-models. Most of them relate to the interface between the engine and the aircraft.

The flight limits should be unambiguously defined:

- When does a flight start and finish and when should the damage calculation start and finish?
- Will damage be counted only between take-off and touchdown?
- Will ground tests be evaluated, and how?
- Will 'Touch-and-Go's' be considered as one flight

(one engine start-up and shutdown) or as many flights as landings?
- Should the damage counting process differentiate a hot start from a cold start?

These decisions are specific to each project and should take into account the hardware and software organisation of the aircraft and engine. A particularly crucial point is the availability of electrical power for measurement and computation when the engine is shut down. Another is the availability of signals from the aircraft. Finally, some physical situations cannot easily be modelled because they are extremely variable. Examples are:

- The decrease in rotor metal temperature after engine shutdown;
- The stress field across a rotor that has not completely cooled before the engine is started again.

Each damage computation model will then be refined and tuned in order to meet these operational requirements.

#### 3.2.2.6. *VALIDATION OF THE COMPLETE MODEL*

One method of validation is to make a comparison between the results from the full model used for life determination and the results from the Reduced Order Algorithms. The procedure consists of computing the damage for each critical area, for a given set of flight profiles including synthetic and real sorties, so that the damage computation can be tested over the whole flight envelope. Though all the models may have been individually calibrated, there can still be some unacceptable over or under-counting. This results from the particular sensitivity of one section of the model to the uncertainty of the previous section. When considering creep damage for a material used near the temperature limit, it is important to remember that a temperature increase of 10°C could halve the creep life. Some experience shows that this can be allowed for fairly easily, because the effect is more significant in terms of temperature gradients and stresses, than in terms of material properties.

| | | |
|---|---|---|
| **Airborne** | *Hardware* | Data Acquisition items<br>(Connections to existing system, new sensors etc.) |
| | | Processing items<br>(Data handling unit) |
| | *Software* | Implementation of the Requirements. (see note below) |
| **Ground** | *Hardware* | Data Transfer medium<br>(Hand held unit, smart card, cables etc.) |
| | | Computer Terminal<br>(Probably commercial off the shelf PC) |
| | *Software* | Implementation of the Requirements. (see note below) |
| | *Data Transfer Procedures.* | |

Figure 9 – System elements to be validated

The reduced model is declared equivalent to the full model when the mean values of computed damage are the same and the maximum difference is below a defined limit. An over-riding constraint is to ensure that the established material properties are not exceeded.

An alternative method for the validation process is to determine the statistical accuracy of the reduced algorithm. This would appear to involve an impracticable large-scale comparison between full modelling and reduced modelling. A method has been proposed (Reference 1) which consists of:

- Measuring the deviations of the individual sub-models (thermodynamic, thermal, mechanical, and material life) from the 'perfect' design model on a set of standard missions;
- Combining the partial results in a Monte Carlo simulation to build a deviation model for the whole system;
- Using the result as a basis for having confidence in predictions for any new missions or design tasks that arise.

The result provides the confidence level for the damage calculation, based on to the uncertainty of the individual sub-models. Currently, this method has not been widely used, but it appears sufficiently promising to be mentioned here.

## 3.3. LONG TERM VALIDITY OF THE REDUCED DAMAGE MODEL

The information generated by the reduced damage model should be kept during the whole fleet life of maybe 30 years. It is thus necessary to keep it up-to-date, despite some inevitable modifications to the engine or the system around it. This section will identify the main situations where a new validation process should be performed. If the reduced model is revised and the ROA's changed then the relevant parts of the validation process must be repeated.

### 3.3.1. IN-SERVICE COMPONENT REPLACEMENT

When a component is replaced during servicing, by another with the same life characteristics, this need only be recorded in the life database. The initial life of the replacement component is 100% if it is new, or is the previously recorded damage if it has been used before. This situation only affects the life database management, and not the monitoring system, except when the monitoring system may need to be reinitialised with fresh values for life-consumed. The need for this will depend on the design of the particular system.

If a component is redesigned, or a modification to other components alters the behaviour of the engine, the rate of damage accumulation for the monitored parts may change. The reduced order algorithms may have to be altered, and

any changes will have to go through the verification and validation process.

Both cases present concerns over the configuration control of the engine standard, the monitoring system standard, and the life database. Control of the correct combination of the first two is essential to ensure that the damage accumulation calculations are correct, and control of the third is essential to ensure that all parts have the correct total damage associated with them.

### 3.3.2. AIRCRAFT AND ENGINES FLY NEW FLIGHT PROFILES

This can occur when introducing the life monitoring system on a fleet-wide basis, or when introducing the system with a new operator. If the new engine usage is clearly different from that already experienced, it is recommended that the accuracy of the reduced modelling for each new flight profile be checked. As the reduced modelling approximation is only valid for the situations where it has been calibrated, a tuning of algorithms may be necessary. Again, it is mandatory that the changes to the monitoring software be correctly managed.

If the changes apply to all engines of the type, it will be necessary to evaluate the need for an alteration to the residual life of all existing parts. This is described in the next paragraph.

### 3.3.3. CHANGES ARE NECESSARY IN THE REDUCED MODELLING

This situation can occur every time a modification in the understanding of engine operating conditions is introduced in the design process and methods. For instance, the following cases should be considered:

- Engine ageing causes component efficiency and internal airflow to change and a new critical zone or a new damage mechanism is revealed.
- A new life design method results in a new predicted life.
- The reduced model has been altered for any reason.

The validation at the end of the development of the new reduced model is a 2-step process:

- Establish the transformation rule, including spread margins, between old and new damage predictions.
- Re-evaluate the residual life of existing components using their actual life, any knowledge about every engine's life, and the new model.

It is essential for the engine manufacturer to have an extensive and continuously updated library of the effective flight profiles performed by the whole fleet. This process is particularly difficult to carry out. In particular, a good compromise between safety margin and customer acceptability can be difficult to reach if the residual lives are significantly reduced.

### 3.3.4. ENGINE LIFE MONITORING HARDWARE CHANGES

These changes can affect the life database or the airborne system. In this case, the normal rules for modifications should apply and any effects on the performance of the monitoring system must be evaluated against the criteria given in paragraph 5. The data that has already been recorded must be kept and must be identified separately from 'post modification' data so that it is available for comparison purposes.

## 4. ENGINE LIFE MONITORING SYSTEM

This section covers the validation of the elements that constitute a practical simplified system. The elements are:

- Airborne system hardware including any new sensors.
- Equipment for data transfer to the ground station.
- Procedures for data transfer.
- Ground station hardware
- Software for data acquisition and handling including the accuracy of the Reduced Order Algorithms in representing the results from the Full Algorithms. This software will always be split into two parts because there needs to be some in the airborne unit and some in the Ground Station. The exact split will vary between projects.

The equipment and processes used to handle the usage data after it has left the station/base where it was generated are excluded.

### 4.1. INTEGRITY REQUIREMENT

Gas-turbine component life-consumption is a progressive and, normally, a long term (1000 hrs. plus) process. Therefore incorrect counting does not cause an immediate problem, and a usage monitoring system is not safety critical in real time in the same way as a flight control system. Also *temporary* counting errors (for a few flights) do not significantly affect the overall situation. However, *systematic* errors, which persist for a large proportion of the time, are a cause for concern.

Monitoring system errors that cause undercounting have possible safety consequences and procedures must be put in place to manage them. On the other hand, over-counting errors do not result in safety consequences but do have economic consequences. These errors will be satisfactorily managed by the procedures used for undercounting.

The incorrect counting may be categorised as follows:

- Inappropriate system design and implementation. This could be due to either incorrect envelope identification, or changes in operational practice.
- System hard faults identified by BITE.
- Subtle system faults, BITE failure or drift, where operation continues but the count is incorrect. An example would be sensor calibration drift.

The first group can produce both temporary and systematic errors. The second group can only produce identified 'temporary' errors that can be corrected and the third group will usually produce 'systematic' errors. All three groups can be managed by the combination of an initial validation programme and an ongoing service review process.

The validation process, which will include all foreseeable circumstances and confirm that systematic errors are very

unlikely to occur during initial operations, will primarily cover the first group. Because operations change through the service life of any engine application, regular reviews of the life control process are carried out as part of current practice. This will still be necessary with a usage monitoring system. However, there will be information, which is more detailed, available from the monitoring system to help the review. Any systematic incorrect counting which escapes these checks would certainly be missed by the present procedures. Therefore any other incorrect counting in this group will be due to *temporary* causes and will not significantly compromise the accumulated damage values.

The second group is the simplest to manage because the fault is clearly identified. A clear and reliable mechanism must be established to ensure that these events are immediately reported to the data management staff so that all identified deficiencies are corrected. Some systems automatically use a pessimistic default value when a detectable fault occurs.

The third group of failures is probably the greatest threat because it is a new type of problem for critical-part life-management, and associated with the introduction of monitoring systems. This type of fault can be effectively managed using the redundancy inherent in the information management infrastructure, which is needed to make effective use of a monitoring system fit. The amount of additional capability depends on the existing techniques and equipment used to manage the fleet, and the information management style chosen by the operator. The ground 'units' can be PC's or mainframe computer terminals or, at the other extreme, databases in hard copy form.

When a system is operated in-service, there must be three elements for practical reasons:

- An 'on-board' unit which gathers the basic data.
- A 'squadron' unit to provide information for first line maintenance.
- A 'headquarters' unit to provide information for spares ordering and operational planning.

Each of these elements can be used to review the results and provide a check on the other elements and will provide an effective means to detect and allow for the third group of failures. In addition, wherever the processing is carried out, it is desirable for the on-board unit to be able to record 1 to 2 flights of raw data for independent processing. This will provide a routine system check, (say once per year) and may be used to investigate suspicious results.

The following examples, in figure 10, provide some quantification of the change in probability of the fleet-wide presence of a (0.38-mm) crack when undercounting occurs. They come from a study on retrofitting RAF Tornadoes with engine monitoring systems. Critical components in the RB199 engine were designed based on the chance of a crack occurring being 'one in 750 with 95% confidence' described in Chapter 6. The table values are considered to be pessimistic scenarios for undetected cases of undercounting and none of the cases generates a significant increase in the overall probability of a crack appearing.

The values in the table are for fleets of 50 and 200. Assume that 5% of the components in a fleet of 200 undercount by 20% for 1/3 of their lives. It can be seen from line 4 in figure 10 that the probability of a crack occurring at the design life rises from 1 to 1.04 in 750.

As a result of these considerations there is no reason, from the usage monitoring viewpoint, for any part of the monitoring system to be safety critical (e.g. DO178B level A or B software). However, other considerations apply to the airborne elements of the system.

| % of cyclic damage or stress undercount | % of life where undercount occurs | % of fleet affected by undercount | Fleet probability change factor (200 engines) | Fleet probability change factor (50 engines) |
|---|---|---|---|---|
| 10 | total | 3 | 1.04 | 1.05 |
| 10 | total | 5 | 1.07 | 1.08 |
| 20 | total | 1 | 1.04 | 1.05 |
| 20 | 33 % | 5 | 1.04 | 1.05 |
| 30 | 33 % | 5 | 1.07 | 1.08 |

Fig 10 – The effect of undercounting on the probability of a crack occurring

The interaction with safety critical control systems, which will generally result in the airborne software being controlled to DO178B level C, is probably the most important. However, it is possible to use Commercial Off The Shelf (COTS) equipment for the ground based elements of the system. It is important to remember that most of the processes, which define the safe lives of critical components, are carried out on COTS computers, both hardware and software. Consideration of the whole system and built in redundancies makes this possible.

## 4.2. SYSTEM ARCHITECTURES

Generally, the data processing for the health monitoring functions is carried out on-board because the results are required at the aircraft, to help the maintenance crew clear the vehicle for the next mission. On the other hand, the calculation processes for usage monitoring need not be carried out on-board. This is because component life is used up slowly during normal operation, during which exceedences, by definition, do not occur. Any significant exceedences will be recorded when each event occurs, and flagged to the ground crew. This can be used to trigger a usage assessment.

The choice of system used on any particular project depends on the specific requirements of that project and the degree of interaction between the 'usage' monitoring and 'health' monitoring functions. For example, an aircraft that is intended to operate for extended periods from remote locations with minimal support will require more

capability in the airborne system.

Any system will require a ground station to act as the interface with the operator's IT networks so that the information from the individual aircraft can be passed to the support functions. Systems where the processing is carried out on-board are frequently able to record data for several flights. For example, the Harrier GR Mk 5 can record usage results for up to eight flights, and record bulk data up to the limit of the storage medium. If bulk data is recorded, ground analysis can later use the data for validation purposes, or investigate incidents in detail. This capability helps to maintain and demonstrate the integrity of the total system and reduces the need for high levels of control over individual items. It may also remove the need for Class A or B software.

With modern commercial capabilities for data storage and transfer, such as smart memory cards and Wide Area Networks, the preferred method for usage monitoring is to record the data on-board and carry out the analysis in a ground station. This approach minimises the airborne element of the system and thereby significantly reduces the recurring costs of the validation and configuration control work required. It also allows the recorded data to be re-analysed later using revised techniques and material information.

## 4.3. SYSTEM ELEMENTS TO BE VALIDATED

The objective of the validation process is to demonstrate satisfactory operation of the system in all foreseeable circumstances.

The process should cover each of the items in the system individually in a manner appropriate to that item. Having produced evidence that each element operates in accordance with the full specification requirements for that unit; further 'end to end' testing is required to demonstrate that the total system operates correctly in all foreseeable circumstances.

The process needs to consider both hardware and software aspects for the additional Airborne Units and for the Ground Station. The major aspects are shown in figure 9.

Note: The Reduced Order Algorithms may be implemented in either the airborne or the ground units depending on the architecture chosen. This affects the software requirements for the two units in any specific case. If these algorithms are implemented in the ground station then the airborne unit software is only required to control the data acquisition process. If these algorithms are implemented in the airborne unit then the basic software in the ground station is only required to handle the accumulated usage counts. However, it is recommended that the ground station should have the ability to process raw data recorded from service to confirm the satisfactory operation of the system and to allow detailed analysis of any changes to the operating pattern.

## 4.4. VALIDATION PROCESS

Validation is the process of ensuring that the whole system works correctly or at least that any deficiencies are known. Figure 11 (also shown in chapter 6) illustrates part of the process: comparing the master and simplified algorithms, and discovering the differences between them. Full validation covers the whole operational process from beginning to end, and should take into account the realities of operating complex systems in adverse conditions. Weather and factors such as the electro-magnetic environment are just two types of adverse condition that may be encountered.

### 4.4.1. AIRBORNE SOFTWARE

The accuracy requirements will depend on the degree of sophistication of the models, both full and reduced. If the models are simple, as was common for engines of the 50's and 60's then appropriate safety margins will be needed. These can be incorporated within the reduced order algorithms, applied to the resulting reference life units, or applied to the authorised lives. For the newest engines, the reduced order algorithms are necessarily adapted to the available memory and precision of the airborne computer.

The validation process is to confirm that the implemented system gives the same output as the full engineering model when 'flown' through each of the full range of missions expected in the service life of the engine. This activity will include using a mixture of missions, some 'synthetic' and some 'real'. The synthetic missions are designed to ensure that all the extremes of the flight envelope are covered with a margin to allow the envelope to be exceeded without causing gross inaccuracies in the algorithms. They should also cover the full range of input parameters and cycle sizes to demonstrate that the reduced models have the necessary robustness. The real missions are taken from recordings of typical flights, either by the actual aircraft or by a similar one and are used to ensure that the algorithms operate satisfactorily with the transient data characteristics of service operation.

Use of an appropriate approval process such as DO178B at the appropriate level depending on the overall architecture of the application is recommended. For DO178B approval would normally be at level C/D but there may be a need to clear it to a higher level if it is closely integrated with other software (e.g. engine control) which itself requires a higher level.

### 4.4.2. AIRBORNE HARDWARE

Airborne hardware is composed of specific measurement devices, the link between the ECS and ELMS, and to the ELM if this is a separate unit.

The airborne hardware is a group of accessory units: speed, pressure and temperature sensors, wiring, avionic unit etc. Their functioning in the environment for the application is validated using the same specifications as other similar units on the engine. This testing would normally include aspects such as the following.

- Functionality;
- Temperature;
- Vibration;
- Electromagnetic Compatibility (EMC);
- Humidity;
- Salt Spray;
- etc.

The functionality and temperature requirements will be

generated by the specific project; the others will usually be covered by specifications such as MIL-E-6041 for EMC and MIL-STD-810 for Environmental Test Methods.

### 4.4.3. DATA TRANSFER HARDWARE

This unit may be both a part of the airborne system and a part of the ground station or it may be designated as a piece of Ground Support Equipment (GSE). The validation process will depend on the system architecture, and the methods used should be appropriate to the status of each item.

### 4.4.4. DATA TRANSFER PROCEDURES

The data transfer process will vary between individual applications but will always be a mixture of automatic data movement between pieces of hardware and the manual activities needed to make this data movement happen correctly. The validation of the data movements can be carried out as part of the programme on the overall system. The procedures which control the manual activities and any techniques used to identify questionable data and to repair it or compensate for it should be subject to audit to provide assurance that they are adequate and satisfactorily maintained.

This audit should take account of and cover the following aspects:

- Identification of personnel carrying out the download under various circumstances;
- Storage capacity of the airborne unit and transfer unit;
- The required frequency of downloads, and the number of aircraft and flights which can be downloaded to the transfer unit, without return to the ground station;
- The identification of, and compensation for, system failures and data loss;
- Programme for regular critical reviews of the system outputs;
- Training programme for operating personnel.

### 4.4.5. GROUND HARDWARE

This will probably be commercially available equipment such as personal computers, and therefore the 'validation' activity is carried out during the selection process used to decide which unit to procure. Any specific testing should be based on the normal practice for ground support equipment used to handle engine and aircraft data.

### 4.4.6. GROUND SOFTWARE

The operating system will probably be commercially available software and therefore the selection process will include assessments of its integrity. Most operating systems are reliable and the basic further constraint is to ensure that it is mature. If the reduced order algorithms are implemented in the ground station then they will be in a block of project specific software that should be subject to testing to check that the algorithms have been implemented correctly. This testing should be carried out on the proposed ground station hardware with the proposed operating system to ensure that the combination functions satisfactorily.

## 4.5. OVERALL SYSTEM

Successful operation of the usage monitoring system requires that all of the components and procedures operate effectively together. Therefore, it is essential that a program of end to end tests be carried out on the complete system after all the individual units have been qualified.

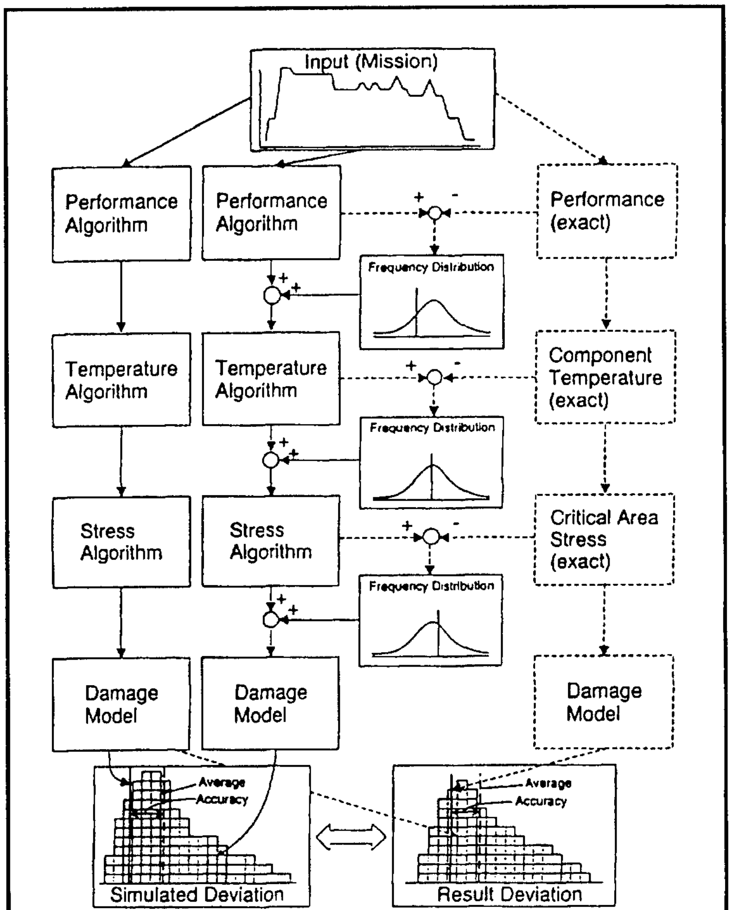



Figure 11 – Correlation between lifing and monitoring results

There is no prescribed formula for these tests and they are normally arranged to suit each specific project. However, a typical practical approach is to carry out a sequence of testing designed to cover the following stages.

1. Demonstrate that each function in the system operates correctly over the full range of input data values covering all possible flight conditions and that all the logic 'loops' and 'switches' in the software behave as intended.
2. Demonstrate that the system operates correctly in real time with typical dynamic inputs.
3. Demonstrate that the system operates correctly in the real environment (EMI, weather, service operation etc.)

### 4.5.1. LABORATORY TESTS

The purpose of these tests is to demonstrate that the whole system functions over the full range of input parameters

and that all the logic in the software operates correctly. They are intended to demonstrate item one of the above list.

The testing requires a representative set-up of the complete monitoring system, where artificially generated input signals can be fed into it and the resulting outputs monitored and checked against the expected values. Ideally, this set-up should include any other units from the aircraft that interact with the monitoring function (FADEC, cockpit instruments etc.). This can be carried out on a powered mock-up of the system, or as a ground test on the complete vehicle. The sequence of input signal values and the expected outputs are defined by the group who produced the Reduced Order Algorithms. The values should be chosen to cover all the possible operating conditions and to exercise all the features of the software (All 'loops' and 'switches'). The values should be chosen to confirm that the boundaries of each loop and switch are correct. For example, if speed ranges of less than 4000 rpm are to be ignored, then the test input values should span and include this value. This controlled 'steady state' testing explores all the combinations but is not able to investigate system performance with inputs that vary in a realistic manner.

### 4.5.2. DYNAMIC TESTS

This testing can be partially done on suitable rigs and bench engines, but can only be completed as part of a flight test programme because of the need for all the inputs to be varying in a fully representative manner. The duration of this test period depends to some extent on the accuracy of the correlation that is achieved, but should not be less than five flight hours.

### 4.5.3. SERVICE ENVIRONMENT TESTS

It is impossible for a practical flight-test programme to cover all environmental circumstances or to reproduce the normal service situation in terms of operating pattern and personnel. Therefore, the results from the initial service phase should be examined carefully over a significant period of flying, say 100 hrs. It should include several aircraft. Initial operation is normally from one base only. As operation starts up at each significant new base, a similar validation exercise should be undertaken.

The exercise should compare the usage calculated by the system with what would be expected for an aircraft carrying out the particular missions. This 'expected usage' will inevitably be a relatively crude estimate of reality. For each project, the important criteria to help in this assessment must be defined. For example, the number of 'Touch and Goes' or, in a helicopter, the number of free-wheel disengagements may be the criterion. The purpose of the exercise is to look for major mismatches between the counts recorded and what is expected from the knowledge of the sortie; e.g. sorties with no counts or sorties with far too many counts.

# 5. EXAMPLES

The following examples are given to explain how two ELMS have been designed to conform to a specific usage specification. They will focus on validation process, ease and cost of operating an engine life monitoring system.

## 5.1. FIGHTER AIRCRAFT: MIRAGE 2000 AND M53-P2

### 5.1.1. SYSTEM DESCRIPTION

The M53-P2 is the engine of the Mirage 2000 multi-role fighter, operated by various airforces. The engine control system uses digital technology. Since 1987, the French airforce has been equipped with a usage tracking and life monitoring system that has three components:

- The ELMS is an electronic unit placed apart from the Engine Control System. It computes the residual life of the critical components on a flight by flight basis. All of the flight and engine parameters are extracted and transmitted by the ECS to the ELMS through a specific connection.
- A ground station, on which the component lives of the whole squadron are managed. This station will periodically be fed with the residual life of surveyed engines.
- A man-operated transfer terminal that makes data transfers between the ELMS and the ground station. This device can used to display the residual life of an engine to the flight mechanic. It also initialises the ELMS after overhaul and reconfiguration of the engine. Its monitoring capacity is forty engines.

After having been initialised with the engine identification and the residual life of the monitored components, the damage calculator works independently of the ground storage system. The lives of twenty critical parts are individually monitored, for creep and low cycle fatigue.

The software is divided into four main sections:

- Thermodynamic, which calculates air temperatures from the measured flight and engine parameters;
- Thermal, which computes temperatures of the solid parts;
- Structural mechanics, which determines the local stress from the temperature and the centrifugal loading;
- Material mechanics, which derives the accumulated damage at all the critical locations from the temperature and stress values. This section integrates the material property database.

### 5.1.2. VALIDATION PROCESS

The following examples will not describe each individual step of the validation process but focus on some 'lessons learned' during the ELMS development.

- During the development of the ROA a very strong sensitivity between creep damage and temperature level was found. Thus, special attention was paid to the precision of the assembled software as measured by the damage. Temperature and stress are not the goals, they are the way to damage and life!
- Numerical approximations due to restricted on-board computer power and memory can lead to unacceptable differences from the results achieved on the ground computer. Extensive testing using various flight profiles is necessary between the ground computer and the on-board system.

Flight testing also led to some discoveries:

- Initially, the real flight-damage count was much higher than expected. This was because the effect of small cycles was over-emphasised. The material property description and the fatigue model were modified, to correctly count small cycles.
- Local temperatures differed from predicted levels in certain flight conditions, and some parts suffered from excessive creep. It was thus necessary to re-identify the full-size model and to further develop the reduced model and re-calibrate it. This type of difficulty can arise whenever an engine is flying a new mission type.

To win acceptance by the operators the hardware and procedures, in particular for data transfer from and to the ground, must be as easy to use, and as secure as possible. In particular this means:

- Any flight monitoring system faults must be displayed to the flight mechanics by a very clear and easy to access flag.
- For operational reasons, the on-board system should be able to monitor several flights before the life information is downloaded to the ground database.
- The downloading device should be able to download and manage data from several aircraft, preferably a whole operational unit as defined by the operating airforce, before the operator has to return to the ground station.

## 5.2. HELICOPTERS: AGUSTA A129 AND GEM1004

This helicopter has an Integrated Multiplex computing system that carries out several functions including aircraft and engine monitoring. The main requirement that led to this configuration was the need to operate for extended periods away from base, with minimal support. Because of this, the on-board system is designed to carry out all the processing and to show the consumed life on the cockpit display screen.

The engine element of this system calculates the low cycle fatigue life consumed for all the critical rotating parts and the creep life consumed for the gas-generator turbine blades. It also monitors the usage of the rating structure and reports the duration and level of any exceedance.

The airborne processing is done in a 'universal' computing system that has flight critical functions. Because it was certified to a high level of integrity, there was no need for any hardware testing specifically for the engine monitoring functions.

There were five elements in the validation programme for the usage monitoring functions.

1. Validation of the simplified algorithms against full stress analysis results. This was to confirm that the simplified algorithms were a reasonable representation of the full cyclic damage analysis using mainframe-computing systems.
2. Validation of the implementation of the onboard units by testing in the systems laboratory, using test cases that exercise all the different routes within the calculation process. This was to confirm that all aspects of the software requirements had been implemented properly so that the system operated correctly in 'non-real time' and with a series of steady state inputs.
3. Flight testing in an aircraft with a data recording capability. This was to confirm that the system operated correctly in the aircraft, in real time with actual engine data inputs and that the other tasks of the computing system did not compromise the engine usage monitoring. The results calculated by the monitoring system were compared with those calculated using a ground system fed with the data from the flight test recorder (approx. 10 hrs of data). The ground system was programmed with the same algorithms as the airborne system. This indicated agreement within 0.5% for the two sets of results.
4. A small sample of the flight test data was re-analysed in an independent ground system with the algorithms implemented by a different software author. This was to confirm that there were no errors introduced via common software authorship of the flight software and the validation software.
5. Periods of service operation on several aircraft were examined to confirm that the correlation between the usage measured by the monitoring system and the known usage of the aircraft was as expected. This was to confirm that operation in service over a wide range of environmental conditions did not cause any significant errors in the calculated usage.

All these elements were applied to the low cycle fatigue counting function because this has a clear effect on flight safety. Turbine blade failure on a multi-engine aircraft does not represent a flight safety hazard and therefore only elements 1, 2 and 5 were applied to the turbine creep monitoring function.

The implementation of the creep monitoring function revealed that for 'continuous accumulation' functions such as this a monitoring system has to operate to very high levels of precision. Because of the need to capture the short periods of very high damage-accumulation caused by exceedences the system has to calculate the damage rate over small intervals (less that 0.5 second). However, the damage rate at normal cruise conditions is very low (approx. 100,000 times lower). The system must be able to register these low damage increments because this rate will apply for the vast majority of the engine operating time.

# 6. CONCLUSIONS AND RECOMMENDATIONS

- All usage monitoring systems should be validated against the manufacturer's design model.
- Each element of the system should be verified against the corresponding element of the master model.
- The implementation must be matched to the users needs.
- Allowances must be made for late completion of such systems. The internal models cannot be completed before there is adequate knowledge of the operational environment, and may need adjustment after the

system has entered service.

- Careful system design can allow monitoring systems to be designed to avoid the requirements of real-time safety critical systems.
- The performance of monitoring systems should be periodically reviewed.
- Consideration should be given to performing life consumption and residual life calculations in ground-based systems. This would facilitate effort, and reduce the cost of implementing updates and corrections in flight-worthy hardware and software.

## 7. SUMMARY

The most comprehensive method of life usage monitoring applies a set of algorithms, to calculate the amount of damage accumulated for each critical component in each engine in the fleet. This chapter describes how the 'reduced order' algorithms suitable for this process are generated and how both they and the system that implements them are verified and validated.

There are three parts to a monitoring system: in-flight systems, ground-based systems and data transfer systems between the in-flight and the ground-based systems.

Validation is a multiple step process that includes ground-based and in-flight testing.

Benefits of a monitoring system include:

- Better knowledge of flight parameters and their impact on component damage;
- Overhaul of critical components closer to the actual life limit with the same degree of confidence in the safety of the process;
- Knowledge of individual engine life-consumption for use with fleet-wide scheduling of inspections, overhauls and aircraft deployment to remote sites.

## 8. REFERENCES

Reference 1 – The Importance of Testing for Successful Life Usage Monitoring Systems. Juergen Broede, Hugo Pfoertner, Klaus Richter. AIMS Conference 98.

Figure 11 was supplied by MTU.

# Chapter 8
## Usage Survey and Mission Analysis

by

*(J. Broede)*

# 1. Usage Tracking

The need for life usage tracking results from the fact that fracture critical parts in aero engines have limited fatigue life, and the desire to operate the engines safely and economically. In this context, *Safe Engine Operation* means ensuring engine integrity by not exceeding the life limits during operational usage. *Economic Engine Operation* means to approach the life limits as closely as possible, in order not to waste life potential. The ideal is to operate the engine precisely to the life limit, and then stop.

"Usage tracking' is a process, which monitors the usage of each individual critical part in terms of life consumption. For critical life limited parts, regulations require that each individual part be traceable throughout its service life history, and that complete records of usage and supporting activities (inspections, repair) are maintained. For this purpose, each critical part is identified by a unique serial number.

Usage tracking would be easy if all missions flown, and all the surrounding conditions were identical. In fact, this is not the case, particularly for military operations, where the diversity of the mission profiles results in significant differences in life consumption.

Early maintenance policies were based on a 'hard-time' philosophy, but the recent trend is more and more towards an 'on-condition' philosophy. This makes it important to know what the 'condition' really is.

There are a number of alternative strategies or methods for estimating life usage. The main methods used are:

- Time Based Usage Tracking;
- Event Based Usage Tracking;
- Mission Type Based Usage Tracking;
- Squadron Based Usage Tracking;
- Usage Tracking Based on 'real-time' measurement and calculation.

As any synthesis more closely represents reality, so the results will become more accurate.

## 1.1. Time Based Usage Tracking

The general idea behind time-based usage tracking is that damage (life consumption) accumulates with time. Both processes (time and damage) are irreversible, and so some correlation should be possible. The simplest type of correlation is the proportional one, saying that in a given time interval a certain fixed portion of life is consumed, and the relationship between time interval and life portion is constant.

### 1.1.1. Different Time Bases

A calendar based tracking method assumes that the engine is regularly used over the years and always with the same mission profiles and mission mixes. The consumed life is directly calculated from the number of years in service multiplied by a 'cyclic exchange rate', which describes the relation between consumed life and time. Once this cyclic exchange rate has been established for a given engine component, life consumption can be easily tracked. The advantage of this method is that inspections and maintenance actions can be accurately planned for long periods.

A flight-time based tracking method tries to correlate the consumed life directly with the flight time, assuming that all engines run roughly as long as the aircraft is airborne. Again, the consumed life is calculated from the time in effective flight hours (EFH), and an appropriate cyclic exchange rate. Now the cyclic exchange rate provides the relation between consumed life and flight time. All that remains is to establish the cyclic exchange rate for all of the monitored areas of the engine.

The aircraft structure itself suffers from fatigue, and requires usage tracking. Life consumption of the aircraft structure is also often related to flight time. Therefore, it may be an advantage that for both engine and aircraft monitoring the same input information can be used, thus avoiding the need to capture two sets of data.

### 1.1.2. Ground Running

The flight-time based usage tracking method does not account for situations where engines undergo ground runs. Ground runs may include, for example, pass-off tests, trouble-shooting and engine installation runs. In this case, engine life is consumed, but cannot be accounted for as the flight time is zero. If engine ground runs are performed regularly, the cyclic exchange rates related to the flight time could be increased accordingly. Another method is to track engine life-consumption based on either the engine operation time or the engine running time. For this, the engine running time, including all flights and ground-runs, should be recorded. As in the previous methods, a cyclic exchange rate must be established, to provide the relationship between life consumption and engine ground-running time.

For the methods of determining cyclic exchange rates, refer to chapter 6.

## 1.2. Event Based Usage Tracking

The general idea behind event based usage tracking is that damage is related to certain events. In particular, fatigue damage is supposed to be directly correlated with stress or strain cycles. Let us assume that stress and strain cycles are directly correlated to load cycles, (ignoring thermal transients and other effects) and that load cycles correlate directly to throttle modulation. Given this, a correlation between throttle movements and life consumption might be established. In practice, it may be more convenient and accurate to measure spool speeds rather than throttle angle.

In this context, a throttle excursion can be considered as an event, but in the same way, a sequence or recognisable group of throttle movements can also be summarised as an event. In the limit, a single flight also can be treated as an event. If it can be shown that the throttle movement sequences are similar for all flights, it may be concluded that life consumption is similar for all flights. If the life consumption of a single flight is known (e.g. from engine design), then life usage tracking can be performed by multiplying the number-of-flights accumulated so far, by the life consumption of the single flight.

As discussed above, for time based usage tracking one could argue that engine ground runs are not covered by this method. They can be, however, included if instead of the number of flights the number of engine runs is counted. In this case the 'engine run' is considered to be the governing event.

Experience has shown that flights might not be as similar as is presumed above. To account for differences between throttle movement sequences, the whole sequence can be decomposed into single throttle excursions. Generally, a flight (or an engine run) contains one main excursion from zero to max back to zero and a number of sub-excursions (e.g. idle - max - idle, idle - intermediate - idle, max - intermediate - max). Now, one could simply count the number of excursions, but this does not lead to a proper result, since excursions with different ranges are known to cause significantly different amounts of fatigue damage.

The solution for this case is to weight the excursions according to their contribution to life consumption. Every range needs an individual weight factor. However, for practical applications it is more convenient to categorise the sub-excursions into major, minor and negligible. The main excursion is normally given a weight factor of one. Negligible excursions are zero-weighted, which means that they need not be counted. Major and minor sub-excursions are weighted with factors between 0 and 1. Finally, the procedure is as follows.

- Count the number of main throttle excursions, including major and minor sub-excursions.
- Multiply the counts by their respective weighting factors.
- Sum the weighted counts.

This results in a number of equivalent main excursions. If the life consumption caused by the main excursion is known, one can now calculate the life consumption of the flight with the analysed throttle sequence.

Another method for determining the life consumption per flight is to record the necessary operating parameters of a representative number of flights and calculate the typical values for all monitored critical areas. Scatter factors and safety factors may be added as discussed elsewhere. This method will probably be more accurate than the decomposition of throttle sequences.

## 1.3. MISSION-TYPE BASED USAGE TRACKING

Generally, military aero engines fly a variety of different mission profiles (also known as mission types or sortie-patterns). The profiles are made up of a number of mission elements such as take-off, climb, cruise high, cruise low, low-speed dash, and any others that may be appropriate. Each mission profile is identified by mission codes or sortie codes. For each of the different mission profiles a typical throttle sequence can be specified. These sequences can be decomposed into individual throttle excursions and an equivalent number of main excursions. The corresponding life consumption per mission can then be determined.

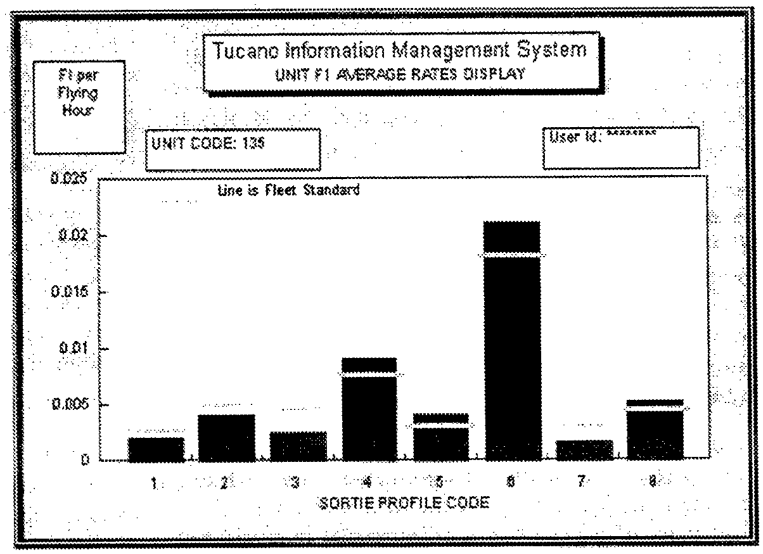



Figure 1 – Comparing squadron and fleet life consumption rates

Mission-type based usage tracking means counting the number of each mission type, multiply them with their respective life consumption per mission and add them up over all mission types. Figure 1 shows an example of mission based usage tracking.

This procedure can be easily extended to account for engine ground runs too. All that is necessary is to determine the life consumption of typical ground runs and include them as 'special mission profiles'.

Mission type based usage tracking is not limited to the event based method but can also be applied in combination with a time based one. In this case, the cyclic exchange rates need to be established separately for each mission type. This can be done either on the basis of well defined synthetic profiles or recorded data from representative missions.

One should be aware that the mission type based usage tracking method is more complicated than the methods where only times or numbers of flights need to be recorded, since here as additional information the mission type is required. This may cause difficulties if some missions cannot be allocated to the defined mission types.

Furthermore, the logistic efforts are higher because more data must be handled and more care is required in order not to mix it up.

## 1.4. SQUADRON BASED USAGE TRACKING

It has been established that aero engine life consumption varies between different squadrons or air bases, even if they fly the same mission types and same mission mixes. Influences include:

- Base locations relative to training areas;
- Differences in aircraft ground handling;
- Varying interpretations of handling instructions;

- The length and altitude of the runways.

To account for these differences one can introduce different cyclic exchange rates or different figures for life usage per flight for the respective air bases. To do this squadron identification must be added to the life usage records.

As more variables such as mission types and squadron are recorded, the logistics and accounting procedures will become more complicated. One may well question whether the improvement in life usage tracking accuracy is worth all of the effort necessary to determine and manage all the different rates and figures.

How cyclic exchange rates, or life usage per flight, figures are determined is discussed in detail in paragraph 2.

## 1.5. USAGE TRACKING BASED ON INDIVIDUAL COMPONENTS

A number of problems associated with the usage tracking methods described above can be overcome if life consumption is calculated individually for each engine or critical component. Because fatigue damage of a critical feature is related to its own stress-temperature history, rather than what is happening in other engines, this approach is the only way of avoiding the premature removal of components.

Algorithms which transform given stress-temperature sequences into consumed life are available elsewhere. The basic idea is to derive the stress-temperature histories at the monitored areas from measured engine operating parameters and convert them into life usage. This can be done, and is, by on-board monitoring systems. This approach covers all of the above differences between different mission profiles and squadrons, and additionally addresses differences between the handling of engines in multi-engine aircraft, actual weather conditions, engine airframe interactions, and pilots.

Of course, the accuracy of the result depends strongly on the capabilities of the algorithms applied. They range from the very simple to the very complex. Simple procedures use equations, which calculate centrifugal stresses, (proportional to the square of the spool speed,) and keep temperatures constant. The most complex procedures take into account transient metal temperature development in the components, centrifugal stresses, thermal stresses, assembly stresses, stresses due to gas pressure, and stresses induced by manoeuvre loads and other influences. The algorithms themselves (irrespective of how sophisticated they are) build a model, and as with every model they are only an approximation to reality. Therefore, the accuracy of the applied algorithms must be analysed and the results taken into account when individual component life-usage-tracking methods are assessed.

### 1.5.1. ON-BOARD OR GROUND-BASED?

Individual life usage monitoring calculations may be performed either in 'real-time' on-board the aircraft or at dedicated ground stations. If the analysis is done on-board then a 'reduced' model of the components being monitored is likely to be used. Such models are required to permit rapid analysis on less powerful computers than could be used on the ground. They may also be required for routine ground based analysis, because taking longer than the flying time to analyse a flight is not an attractive proposition. If a 'reduced' or simplified model is used for analysis it must be validated against the full design model, which includes the full component and materials databases.

It has been suggested that life usage data may be transmitted to the ground, and this technique is used for trending by airlines such as Lufthansa. However, the volume of data required for life usage monitoring a fleet of engines would be very high. If the data is to be analysed at a ground station the raw data must be recorded and transferred from the aircraft. If the analysis is done on-board then only the results need to be transferred on a routine basis. Prudence would suggest that a bulk-data storage-device should be available for trouble-shooting for special investigations and for periodic re-validation of any 'reduced' models that may be in use.

During the development of this report PC development has progressed rapidly and it is reported that modern ground stations can now analyse complete flights in about 15 - 20% of flight duration.

### 1.5.2. BULK DATA STORAGE

When raw data is required for analysis, the required engine operating parameters must be recorded, and the recorded data transferred to the processor system, which will perform the calculations. Data from one or more flights may be recorded on a single data carrier. For proper data evaluation, it is necessary that the recordings are complete, i.e. that they contain the whole period from engine start to engine shutdown. If data from more than one engine run are stored together, the engine runs should be clearly separated to avoid misinterpretation and faulty results.

Signal selection should be specified according to the intended further use. As a minimum, all operating parameters that are required in the subsequent evaluation process must be included. In many situations, it will be beneficial if additional information is also available. When signal sources are selected the accuracy and quality of the input signals should be considered. As with every measured quantity they deviate from the physical 'truth'. These deviations need to be considered, too. Details of these points are discussed elsewhere.

The bulk data can be evaluated with different intentions. One intention could be the usage tracking of individual engines. In this case, a clear tagging of the data, which at least uniquely identifies the engine to which the data belong, is required.

Furthermore, it might be important to evaluate the flights in the same sequential order as they were flown. This will be the case when damage increments depend on the current state of life consumption (e.g. when crack propagation life is used). The order of the recordings should be tagged accordingly, either by a unique identification of the engine run number or by date and

time.

If the intention is to establish cyclic exchange rates (e.g. for the use in time-based tracking methods), then it is sufficient to tag the recorded data only with the engine configuration (e.g. engine type and engine variant codes). This may be supplemented by information indicating the mission types flown and the air base, if required.

A pool of bulk data is very useful for the design, improvement and verification of life monitoring algorithms. Experience shows that tests of newly developed or improved algorithms are much more effective if recorded data are used, rather than synthetic data. In particular, it can be checked rigorously to ensure that the control structure of the system (sensing engine start and engine shut down, switching between different modes) works properly. The criteria for signal plausibility checks and the restoration models for substitution of faulty data can be fine-tuned with real data, which is virtually impossible if only synthetic data are available.

## 2. CYCLIC EXCHANGE RATE

### 2.1. DEFINITION AND DISCUSSION

The Cyclic Exchange Rate (also called Mission Severity Factor or simply β-Factor) describes the relationship between life consumed (mainly fatigue life under cyclic loading) during operational usage and the correlated time interval. As discussed above, the time-base can be calendar time, engine flight time or engine running time. Certain types of events (such as flights, single throttle movements, sequence of throttle movements) are also in use as a 'time-base'. For most military applications, the engine flight time in hours (EFH) is used as the base, whereas civil applications rely on the number of flights (often called 'cycles'). Nevertheless, the general description applies to all types of time-base.

The term 'cycle' takes a number of different meanings, including the following.

- Flight;
- Single throttle excursion from a certain beginning level to a target level and back to the beginning;
- Sequence of throttle movements;
- Closed loop in the stress-strain history at a certain critical area;
- Reference cycle, meaning the most damaging loop in the stress - strain history at a certain critical area under the design reference mission.

In many cases, the correct meaning of the expression 'cycle' can only be seen from the context. If the word 'flight' is taken as the meaning of cycle then this also has many meanings including:

- Engine-on to engine-off, but only if the aircraft flys;
- Take-off to landing;
- Take-off to landing, excluding 'Touch and Go' manoeuvres;
- Whatever the operator says a flight is.

The most highly stressed areas of a part are it's so-called 'critical areas'. Generally, the most highly stressed area or feature determines the life of an engine, component, or part. The life consumed at a critical area depends on the operational usage of the individual engine. In general, life consumption is the consequence of the variation of stress and temperature, and the only physically correct method for life consumption determination is the assessment of the stress-temperature histories at the critical areas. It is quite natural, that under variations in operational usage the stress-temperature histories will also change, resulting in changes of life consumption. However, this effect is not proportional for all critical areas. In fact, some engine manoeuvres selectively damage some critical areas, while other critical areas are virtually unaffected. For example, Quick Reaction Alert starts cause extreme life consumption in bore regions due to the high thermal stresses induced by the steep temperature gradients over the disc. However, hot re-slams are typically detrimental for rim areas due to the high thermal stresses induced by the 'inverted' temperature gradients.

These simple examples make it clear that *the* life limiting area of a component does not exist, and cannot be identified from the design mission alone. Consequently, one has to consider a number of critical areas as candidates for becoming the life-limiting feature under different patterns of operational usage. From the foregoing, it is clear that life usage monitoring should be applied to all of the potentially life limiting areas of an engine part. This includes the calculation of cyclic exchange rates when these are used.

To determine cyclic exchange rates on a sound basis, one needs a pool of real (i.e. either directly evaluated or flight recorded) flight data. This pool should include a representative number of engines and a representative number of missions per engine. The spectrum of missions per engine should encompass all mission types flown in a typical mix. However, one should be aware that it is much more important to have data available from a representative number of engines than to have many data from only a few engines. This can be concluded from the following consideration.

Life consumption is a cumulative process. This means that the consumed life at an individual critical area of a particular part will be summed up over all flights of the part's service time. Quite naturally, over this long period, the critical areas will experience the whole variety of mission types and manoeuvres with high and low damage rates, probably including the extremes. However, the extremes will be rare, and provided the aircraft will be operated typically, the deviations will be balanced out over the part's life. It can be concluded that the average value (i.e. cumulated life consumption divided by cumulated time) will be sufficient to describe the life consumption of an individual critical area of a particular part. (Provided that the 'typical' operational usage will not change over the entire service period). This individual engine-specific average value can be interpreted as an 'individual' cyclic exchange rate.

In contrast, different engines will respond differently to the same operational usage. The reasons are manifold. Most important for our considerations is that these differences do not balance out over the fleet's service

period. This is shown in chapter 9. Thus, a more detailed statistical treatment is necessary. The target should be to derive a 'worst case' or 'safe' cyclic exchange rate. Therefore, the fleetwide statistical frequency distributions of the exchange rates of each critical area should be established and a reasonable quantile (say 95% with 90% CL) determined. This quantile will be a figure somewhat higher than the fleet average for each critical area.

The factor between the 95% - quantile and the average of this distribution can be considered as a dispersion factor for the cyclic exchange rate. Once this dispersion factor has been established, a 'safe' cyclic exchange rate can be directly derived from the average over all individual cyclic exchange rates. This may be of particular interest if algorithms or operational procedures change and new cyclic exchange rates need to be predicted.

It might happen that the number of individually monitored aircraft is not large enough to derive the dispersion factors individually for each critical area. In this case, it could be assumed that the dispersion factors are equal for all monitored areas. Based on this assumption, the statistical distributions for all individual exchange rates can be normalised (preferably with their average values) and combined to a single value. Assuming a number of 30 to 50 monitored critical areas per engine and a pool of 10 individually monitored engines of a fleet, one will achieve a statistical frequency distribution with 300 to 500 samples. This will be sufficient to derive a common dispersion factor for the 95% - quantile with 90% CL.

The 95% - quantile of the cyclic exchange rate distribution determines a 'safe' cyclic exchange rate which for 95% of the engines is greater than or equal to the actual individual life consumption, i.e. for these engine a conservative approach. On the other hand, 5% of engines really do consume more life than predicted by the 'safe' cyclic exchange rate. This results in some residual risk. This residual risk can be quantified as the probability that life consumed in excess of the released limit coincides with a 'weak' part with an actual strength close to the -3 sigma design line. One can easily see that this probability is very small. Thus, the risk can be categorised as remote. However, as a statistician once said 'Statistically speaking, if it can happen, it will happen. It is only a matter of time!'

## 2.2. PROCEDURE TO ESTABLISH CYCLIC EXCHANGE RATES

From the consideration above one can derive a recommended procedure to establish 'safe' cyclic exchange rates, as below.

- Collect real (i.e. recorded) flight data from a statistically representative number of engines with a sufficient number of flights per engine;
- Calculate the life consumption increment of each monitored critical area individually for each engine and each flight. The calculation process should be based on a thermo-mechanical model including all contributing transient temperatures and stresses. Life consumption should be expressed in physical units (preferably in multiples of reference cycles per engine run).
- Calculate the feature specific individual cyclic exchange rate, i.e. the average life consumption per unit time for each individual critical area for each engine. To do this, sum up all life consumption increments of the critical area in question and divide by the accumulated time (for the possible different time-bases refer to the discussion above). The frequently used process of *calculating the life consumption per unit time and then averaging these figures* cannot be recommended, because this process contradicts the physical nature of damage accumulation.
- Establish a statistical frequency distribution of these feature-specific cyclic exchange rates for all engines. Determine the 'safe' cyclic exchange rate as an acceptable quantile (with a suitable confidence level) of this distribution. Repeat this step for all monitored areas.
- If the number of engines within the investigated pool is not large enough to establish valid distribution curves then calculate the average of individual cyclic exchange rates (i.e. the average of the feature specific averages). Then normalise the individual cyclic exchange rates with this average value.
- Combine the normalised figures of all monitored areas to a common distribution curve.
- Determine the dispersion factor between desired quantile and average of this common distribution.
- Multiply the average of the individual cyclic exchange rates (which can be interpreted as an 'average' cyclic exchange rate) with the dispersion factor to yield the 'safe' cyclic exchange rate for each monitored area.

The desired quantiles may be determined by fitting the data to some commonly used distribution law (for example Weibull-Distribution or Normal-Distribution).

A fleetwide investigation of the RB 199 engine in the GAF MRCA Tornado fleet has shown that the dispersion factor for cyclic exchange rates typically takes values of about 1.3.

# 3. DATA SELECTION CRITERIA

## 3.1. PARAMETER SELECTION

Life usage calculation and tracking will require a variety of input parameters. Which parameters are needed will depend strongly on the method used. If, for example, the chosen usage tracking method is simply time-based, then only the flight time or engine running time needs to be recorded. In contrast, for highly sophisticated 'real-time' on-board monitoring systems a long list of input parameters may be necessary. The requested data can be categorised in 4 groups (the following list is considered indicative and certainly not exhaustive).

- Engine parameters that include:
  a) spool speeds, shaft speeds;
  b) engine intake temperature;
  c) compressor exit temperature;
  d) stator outlet temperature;
  e) turbine blade temperature;

f) exhaust gas temperature;
g) engine intake pressure;
h) compressor exit pressure;
i) jet pipe pressure;
j) oil temperature;
k) fuel flow;
l) afterburner-select indicator;
m) pilot's lever demand.

- Aircraft parameters that include:
  a) air temperature;
  b) indicated airspeed;
  c) pressure altitude;
  d) angle of attack, angle of side slip;
  e) g-load, normal acceleration, yaw angle velocity;
  f) weight-on-wheels indicator.

- Configuration data as:
  a) aircraft type, aircraft variant;
  b) engine type, engine variant;
  c) aircraft and engine serial numbers;
  d) monitoring system hardware and software versions.

- General information:
  a) date and time;

It is recognised that a strong correlation exists between several of the above parameters. In such cases, the correlation can sometimes be used to derive other parameters thus making some direct measurements unnecessary. A general rule for engine life monitoring should be to use those signals already available for other purposes, e.g. for engine control, cockpit display or crash data recording. This rule is virtually mandatory when existing aircraft-engine configurations are to be equipped with monitoring systems, but is well worthwhile for new developments. Experience has shown in many cases that it is much easier to derive a required parameter from other measured data with a suitable model algorithm than convincing customers and partners that additional sensors, and data conditioning devices are necessary. Particularly in military applications, every piece of equipment not required saves weight and cost. An exception may be when existing instrumentation does not cover the operating envelope of the monitoring system with sufficient accuracy. However, there are examples where problems with poor quality sensor-signals have been solved with appropriate algorithms in the monitoring system software.

| Signal | Accuracy | Resolution |
|---|---|---|
| Spool speed | 0.1 % | 0.05 % |
| Temperature | 1 K | 0.5 K |
| Engine intake | 4 K | 2 K |
| Compressor exit | 4 K | 2 K |
| Stator outlet | 4 K | 2 K |
| Exhaust gas | 2 K | 1 K |
| Turbine blade | 4 K | 2 K |
| Oil pressure | | |
| Engine intake | 2 kPa | 1 kPa |
| Compressor exit | 10 kPa | 3 kPa |
| Jet pipe | 5 kPa | 2 kPa |
| Indicated airspeed | 2 kts | 1 kts |
| Pressure altitude | 100 ft | 50 ft |
| G-load | 0.01 g | 0.005 g |

Table 1 - Typical Signal Requirements

## 3.2. SIGNAL QUALITY

The signal quality is mainly determined by the characteristics of the sensor and signal acquisition chain. These characteristics include accuracy, resolution, possible drift, and time delays in the response to transient inputs. From the monitoring point of view, the requirements for the sensor characteristic should be in most cases no more stringent than from other clients. If higher demands are considered, the necessity should be carefully analysed. Typical requirements, normally sufficient for life usage monitoring, are given in Table 1.

Another important quantity is the signal sampling-rate (also called update rate or iteration rate). The frequencies need to be high enough to detect all modulations of the signal and to avoid aliasing errors. On the other hand, sample frequencies at too high a rate should be avoided because they do not provide additional or more accurate information. The optimum sampling rate may depend on the component that is to be monitored. Typical sampling rates are shown in Table 2.

| Monitored Component | Sampling Rate |
|---|---|
| Compressor discs | 1 ... 2 Hz |
| Turbine discs | 1 ... 4 Hz |
| Shafts | 1 ... 2 Hz |
| Turbine blades | 2 ... 8 Hz |

Table 2 - Typical Signal Sampling Rates

Note: Aliasing error means an error introduced by a sampling measurement system if the sampling rate is not at least twice the highest frequency component of the signal being measured. It prevents determination of the frequency of the signal, and can result in a measurement offset.

In all cases the verification and validation processes described in chapter 7 will allow the system sensitivity to sampling rates, accuracy and resolution to be measured before the first hardware systems are specified and built.

## 3.3. PLAUSIBILITY CHECKS

Monitoring input data are, as are all measured signals, sensitive to disturbances. To ensure that lifing calculations are reliably executed, a number of checks should be performed to detect possible implausibility of the input. Typical checks are range, rate, cross, and model checks.

*Range check*: for every signal, a plausible range is established within which all measured data should fall. The plausible range is derived from physical limitations (e.g. negative pressures are impossible) and operational limitations (e.g. spool speeds beyond burst speed). Data outside the plausible range are declared as implausible.

*Rate check*: for every signal, the change between two successive samples is scrutinised. If this change exceeds a plausible value, the data are declared as implausible. The

plausible range for the data change is established based on the engine and aircraft characteristics. (E.g. maximum and minimum changes of spool speeds are derived from engine acceleration and deceleration capabilities in the considered aircraft-engine configuration).

*Cross check*: can be performed if for a certain signal more than one source exists. In this case, the deviations between the different sources are considered. If the deviations exceed a predefined range, the signal can be regarded as implausible. However, it is also possible to employ a more sophisticated logic to detect which of the sources produces the implausible input.

*Model check*: works in a similar way as the *cross check*. In this case the second source (or further sources) for the signal are model calculations which use other signals as input. The models describe the correlation between different signals. The models should be established on measured data, which are unambiguously plausible. Again, a range of acceptable deviations between the measured and modelled signals is to be defined. Deviations exceeding this range are an indicator that signals are implausible. If it is necessary to identify which of the sources produces the implausible input, again a more sophisticated logic is needed.

A strategy for handling implausible data must be decided and built into any automated life usage monitoring equipment. All detected cases of implausibility should be reported to the user and appropriate warning messages included in the output.

### 3.4. RESTORATION OF FAULTY DATA

If faulty data are detected (if data are declared as implausible) then a process will be necessary to restore the information. Depending on the extent of implausibility, a variety of measures can be taken, ranging from simple interpolation to sophisticated substitute calculation techniques.

If, for example, for a generally reliable signal only a few samples in the data stream are implausible and the preceding and following samples plausible, then linear interpolation between the plausible values is sufficient. The error resulting for the life usage calculation is negligible in such a case.

Longer periods with an implausible signal or total loss of an input source will require substitute calculation of this signal. This can be done with a model using other signals as input. The same models as used for the model check can be used, but consideration should be given to adding some offset in order to push the life usage results to the conservative side. When establishing such models, the effects of such substitutions on the life-usage results should be carefully investigated. The results of this investigation should be assessed to determine the maximum possible errors.

Input data may be even more disturbed. If more than one signal breaks down simultaneously, substitute calculation could produce unacceptable errors or even might become impossible. In such a case, life usage calculation should be interrupted or the lifing result declared as implausible. Implausible life usage results should be discarded and replaced by substitute results which may be based on the methods for time based usage tracking or event based usage tracking. For this purpose, it is worthwhile recording further engine operational data such as engine running time, engine flight time, number of engine runs or engine flights. Depending on the complexity of the engine life usage monitoring system, substitute calculations for filling the monitoring gaps can be done automatically in the on-board processor or be performed in the ground station. Although these simple methods are crude, one can argue that the overall error in life usage tracking is acceptable as long as such actions are rare and remain the exception. In any case, a substitute calculation - however crude it may be - is still better than leaving the gaps unfilled.

### 3.5. SELECTION OF FLIGHT POINTS

Techniques for *engine-health*, rather than *life-usage* monitoring, use selected flight points for their analyses. The emphasis is on stabilised flight and engine running conditions, which are assumed to occur in certain phases of the flight, and which are a prerequisite for performance trending analyses.

For life usage analyses, these stabilised conditions are of little interest. This is because life consumption (in particular fatigue life consumption) is mainly caused by changes in the engine running conditions, and those flight points where the critical areas experience extreme stresses and temperatures. For 'well defined' mission profiles - as they appear in commercial aviation - these phases of an engine run are well known. They encompass:

- Engine start;
- Engine run up for aircraft take off;
- Cruise climb;
- Holding pattern;
- Landing and thrust reverse;
- Engine shut down with the subsequent cooling phase.

Commercial considerations about passenger comfort, satisfaction and repeat business ensures that all flying is done as smoothly as possible. However, military mission profiles look completely different. They are composed of continuous sequences, often of relatively violent manoeuvres. This makes it extremely difficult to predefine simple flight phases that produce information necessary and sufficient to accurately derive life consumption. One should be aware that the different types of critical areas react differently to various manoeuvres, so that flight phases important for one area may be negligible for another one.

Consequently, the only technique that can be recommended for gathering flight data is continuous recording with a sufficiently high sampling rate. Pre-definition of criteria to select relevant flight points is very difficult and includes the risk of missing important information with the possible consequence of non-conservative life quotation.

#### 3.5.1. DATA COMPRESSION

It must recognised that there are strong pressures to reduce the amount of data to be handled and transferred. One

measure could be to reduce the signal sampling rate, but there are limitations. Other options are data compression techniques. These can be categorised as 'Lossy' and 'Non-Lossy' or non-reversible and reversible. Non-lossy techniques allow a complete restoration of the original data during the decompression process, whereas the restoration can be incomplete when lossy techniques are used. In less agile aircraft, such as civil airliners some trending systems record steady state information together with a duration field that shows the period for which the steady condition was maintained.

Experience with standard non-lossy data compression techniques applied to recorded flight profiles of military missions shows that compression factors of 10 or 12:1 are typical. Higher compression rates can be achieved with lossy techniques; but these cannot be recommended for military mission profile data, since they bear the risk of losing important information.

### 3.5.2. TAGGING OF DATA

For life-usage monitoring, it is necessary that data are not confused or mixed up. Thus monitoring data must be uniquely assigned to an individual engine, engine component or engine part. To fulfil this requirement, monitoring data are tagged at least with aircraft and engine identifiers (e.g. serial numbers), date and time (or - alternatively - with a unique engine related count of engine runs).

Additional data should be added if they are demanded for special evaluation. These additional tagging data may include engine configuration, mission type and airbase identifiers, as well as special comments advising about irregular conditions.

# 4. DATA TRANSFER

## 4.1. FUNCTIONS AND REQUIREMENTS

Engine monitoring systems are built up as distributed systems, as shown in figure 2. They consist of components located in the aircraft and at the engines, and ground equipment. Reliable and accurate communication between these system parts is essential. Simple systems (which only collect data on-board) require only unidirectional data transfer, i.e. download of recorded data into the ground station for further processing. More sophisticated applications need regular bi-directional communication. Data transfer from ground systems to the on-board components encompasses:

- Initialisation of on-board equipment with configuration and current usage data after new installation or repair;
- Corrections in cases where substitute calculations were necessary;
- Diagnostic queries;
- Updates to rules.

Data transfer from aircraft to the ground station consists of:

- Recorded flight profiles;
- Life usage results;
- Additional operational data;
- Diagnostic information and warnings;
- Tagging information.

Decisions must be made, about which tasks are to be performed on-board and which in the ground stations, at an early stage of design and development. These decisions generally determine the interfaces and data transfer requirements. Insufficient planning of the data transfer facilities can jeopardise the success of the whole system or - in less severe cases - cause additional cost.

Data transfer should be carried out with a high level of automation. Manual support or even manual transcription of data should be kept to a minimum. Provisions should be made to ensure that data integrity is always maintained.

## 4.2. EQUIPMENT AND MEDIA

Data transfer equipment is the means to perform communication between aircraft installed parts and a ground station. Basic requirements are that data transfer must be quick, easy to operate, robust and user-friendly. Depending on the architecture of the whole monitoring system and the amount of data to be transmitted, the transfer equipment can make use of removable memory devices (such as cassettes, tapes, discs and solid state modules) or data-bus links.

It is desirable for logistics reasons, that ground equipment and data transfer devices are common for different types of aircraft and engines. This requires a high degree of flexibility in order to cover the entire range of possibly different requirements and formats of the different aircraft and engines. It is also difficult to achieve when procurement is on a project basis, and aircraft systems are purchased from a single prime supplier through competition.

In many cases, quick-look capabilities are required, particularly for NO GO indications or warnings. They either can be realised as on-board displays or can be provided as functions of the data transfer device. The main technical issues here are the availability of the data at a suitable point. If a rapid aircraft turn around is required, without the pilot leaving the cockpit, then it may be necessary to provide more than one access point for quick-look displays.

# 5. RECORDS AND ACCOUNTING

Records and accounting - in this context - means the activities necessary to record the life usage data of engine parts, engine components, modules, and complete engines in a central or distributed register, and to keep these records always up to date. Generally, the accounting activities are ground-based tasks and are performed within the ground equipment.

To ensure correct accounting of lifing data, a well organised parts management system is required. Such a system may range from a simple card index system to complex computer-aided information management system, which is able to communicate with airborne parts of the monitoring system directly via data transfer equipment.

The database management activities should be well

integrated into the maintenance process and tailored to the overall maintenance philosophy and the logistic requirements. The following information can be provided:

- Monitoring of fleet-wide life usage (individual engine and components, fleet average, frequency distribution);
- Projecting of engine overhaul;
- Provisioning of spare parts and logistics support;
- Component life matching during engine builds, as shown in figure 3;
- Correlation of maintenance history with operating conditions (mission profiles, life consumption, operating environment).

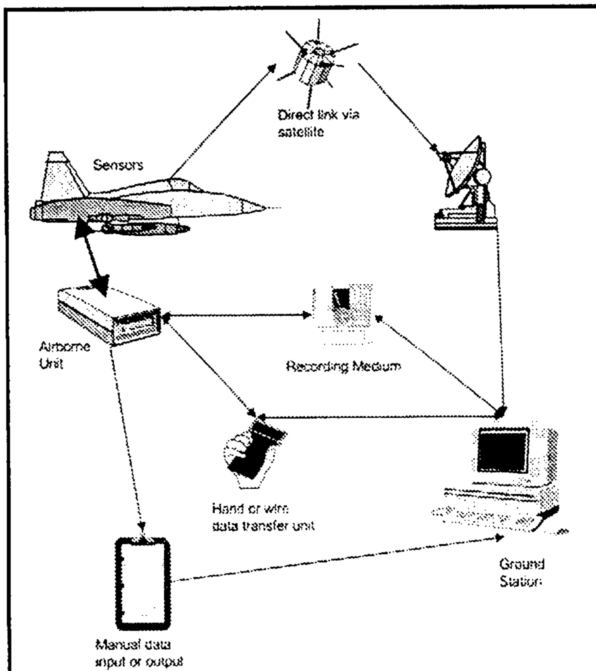


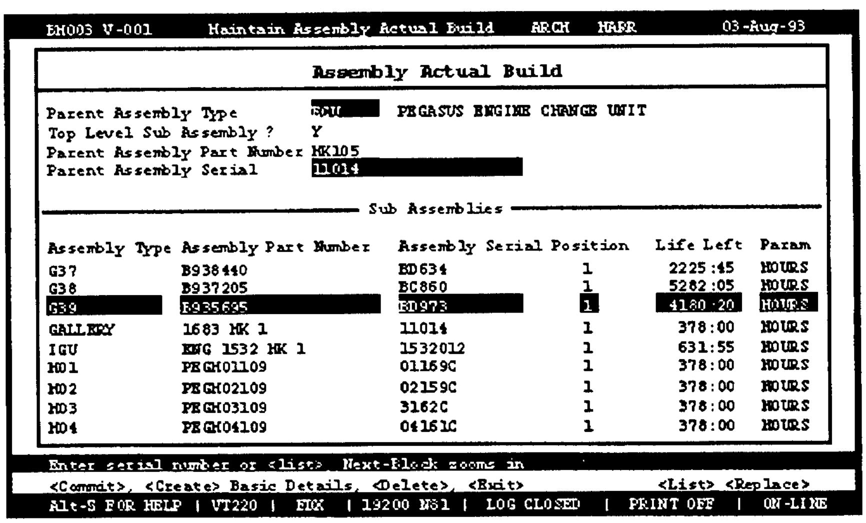

Figure 2 – Typical data transfer methods

## 5.1. Logistic Systems

Maintenance of any type of configuration data requires logistics information. At the very least, this will encompass keeping track of which engines are on which aircraft. More sophisticated systems with full parts tracking capability will require regular updating of part numbers and serial numbers for all tracked components, modules and engine parts.

Updating of configuration data is necessary when:

- A new aircraft is received or an old one is moved to another location;
- An engine is removed or replaced;
- An engine is disassembled or rebuilt;
- Parts of the engine monitoring system hardware are exchanged;
- The engine monitoring system software is updated.



EH003 V-001 Maintain Assembly Actual Build ARCH HARR 03-Aug-93

Assembly Actual Build

Parent Assembly Type GCU PEGASUS ENGINE CHANGE UNIT
Top Level Sub Assembly ? Y
Parent Assembly Part Number MK105
Parent Assembly Serial 11014

Sub Assemblies

| Assembly Type | Assembly Part Number | Assembly Serial | Position | Life Left | Param |
|---|---|---|---|---|---|
| G37 | B938440 | BD634 | 1 | 2225:45 | HOURS |
| G38 | B937205 | BC860 | 1 | 5282:05 | HOURS |
| G39 | B935695 | BD973 | 1 | 4180:20 | HOURS |
| GALLERY | 1683 MK 1 | 11014 | 1 | 378:00 | HOURS |
| IGU | ENG 1532 MK 1 | 1532012 | 1 | 631:55 | HOURS |
| M01 | PEGM01109 | 01169C | 1 | 378:00 | HOURS |
| M02 | PEGM02109 | 02159C | 1 | 378:00 | HOURS |
| M03 | PEGM03109 | 3162C | 1 | 378:00 | HOURS |
| M04 | PEGM04109 | 04161C | 1 | 378:00 | HOURS |

Enter serial number or <list> Next-Block zooms in
<Commit>, <Create> Basic Details, <Delete>, <Exit> <List> <Replace>
Alt-S FOR HELP | VT220 | FDX | 19200 N81 | LOG CLOSED | PRINT OFF | ON-LINE

Figure 3 – Component life matching during engine assembly

As a minimum, manual entry of logistics data must be possible. A more effective means would be an electronic data transfer via appropriate media (e.g. tapes, diskettes) or data links. Logistics data will be required from different sources as well as transmitted to different destinations (e.g. engine manufacturer, overhaul facilities, other airbases).

In many cases, a central database is employed. If this is the case, the ground station should be capable of both uploading and downloading information stored on the central database. It may be additionally be required that certain other engine monitoring data can be transferred to the central database.

Engine removal and installation is logged in order to ensure the validity of the engine configuration database. Maintenance actions to remove and replace life-tracked parts are logged, too, and it may be a requirement that the status of maintenance actions is tracked at a ground station until the respective action is closed.

If separate systems have been established for aircraft and engine maintenance, then the engine monitoring ground station should not require duplicate entry of information. The engine monitoring ground station should be integrated with other aircraft maintenance-data systems, if these already exist. Configuration data should be shared for both systems.

## 5.2. Configuration

During the life of an aero engine, it will be removed from service several times for overhaul and repair. In the course of these maintenance activities, parts are often replaced with spare parts, where the parts may be - although interchangeable - of a different standard. The same is true for complete modules from modular engines. In both cases, a high degree of interchangeability between

different standards is required.

This interchangeability must be reflected in a configuration control function within the parts management system. Its main task is to record the exact configuration of engines. Less obviously, it should also manage the configuration of the engine monitoring system itself, which also will undergo changes and improvements.

Modifications lead, in many cases, to improvements of the lifing behaviour of the affected parts. This improvement means that either the cyclic life of the improved standard is higher than of the original part, or that less life is consumed during a given mission profile. Both these possibilities need to be adequately addressed in the monitoring system. The consequence is that the life usage tracking function must be informed about the physical engine configuration. In particular, if life consumption is monitored in an on-board system, the configuration information must be evaluated on-board to select correct algorithms and constants.

Additionally, compatibility checks are required. Checks are required to ensure that the installed monitoring-system hardware and software are able to correctly monitor the actual engine configuration. Any detected mismatch must be flagged and corrective actions taken. Provisions are also required to protect the life usage data against incorrect updates, and to provide the data required for substitute calculations.

## 5.3. DATABASE SYSTEMS

A database should be incorporated in the ground station. This database should contain all of the detailed information necessary for life usage tracking of each engine belonging to the squadron or fleet. This information should include a complete description of the engine configuration, serial numbers of engine, modules and life-limited parts as well as the life usage data for all the parts. Additional information, such as accumulated counts of engine runs and engine flights, engine running time and engine flight time or periods of special conditions (e.g. afterburner or thrust reverse selected), may be helpful. This may be used for further analysis, trend interpretation and statistical evaluation.

The content of the database should be regularly updated as and when engine life-usage data of the aircraft become available. The preferred frequency is a daily update. If aircraft are to operating away from base then provision must be made to collect and store data while they are away. When the aircraft return, the data must be correctly added into the database.

Two different ways of storing life usage data in a database are in use.

- Storage of the total number of cycles accumulated to this point in time;
- Storage of the life usage increments for each engine-run.

The UK Harrier Engine Monitoring System (EMS) stores results on a flight-by-flight basis for up to eight flights, while the GE Tornado OLMOS system stores the cumulative total. This reflects the different operational scenarios for which the aircraft were designed, and different design philosophies. The ground storage systems both store the data on a flight-by-flight basis.

If the on-board and ground systems store the results in different ways checks are needed. If in such a situation the on-board algorithms do not check the life usage results prior to an update, then the necessary plausibility checks should be performed at this point in the ground station.

If life usage calculations performed in the ground station are based on recorded mission profiles, then the usage increments are directly obtained. For storing accumulated life consumption, the increments simply need to be summed up. Although storage of accumulated life with updates overwriting the previous values uses the storage capacity more efficiently, it seems worthwhile keeping the life counts per certain dates, e.g. twice a year. This allows for trend analysis, in particular for indications whether there are changes in operation or other factors influencing the life consumption. The diagrams in chapter 9 provide an illustration of cumulative life usage data.

The only reason for having a ground station is to extract information from the database. This encompasses:

- Calculation of cyclic exchange rates;
- Substitution and restoration of lifing data;
- Trending;
- Conversion of cyclic data into time based data;
- Reading across from monitored parts to parts not directly monitored;
- Preparation of initialisation data for the on-board system.

Cyclic exchange rates provide the relationship between cyclic life consumption and a time-base, which describes the period within which the life is consumed. The time-base can be either the engine flight time or the engine running time. The cyclic exchange rate is the ratio of the life consumed (in terms of cycles) divided by the corresponding period (in hours). Cyclic exchange rates can be provided for:

- Individual engines;
- Certain mission profiles;
- Squadrons;
- Air bases;
- Fleetwide averages.

Safety factors should be included for deviations and scatter in all cyclic exchange rates. As mentioned earlier fleet averages are probably not the safest fill-in factors to use.

Substitute calculations should be performed if lifing data are corrupted or lost. It is mandatory that reliable backups exist, so that data corruption or loss influences only short periods. Substitution values for these periods can be calculated by using cyclic exchange rates and the length of each period.

Trending of lifing data means both keeping track of historical data and projecting trend lines into the future to anticipate the need for action. Trends are normally provided for individual engines, but they can also be

evaluated for certain mission profiles, air bases or fleet-wide. Trending data are often not required in terms of cycles but in terms of time. Therefore, it is usual to convert remaining life (in cycles) into remaining life in engine hours using cyclic exchange rates. With the daily engine usage in hours, one can forecast necessary activities for each engine or aircraft by calendar dates. Evaluation of trends produces information about changes in usage and isolated events that can be correlated with reported problems and diagnosis.

For engines where not all life-limited parts or critical areas are directly life monitored, it is possible to estimate the life consumption of the parts concerned from the areas directly monitored. It is usual to apply read-across factors between the parts or areas concerned. But when doing so one should be aware that read-across factors may vary in the same manner as cyclic exchange rates, i.e. they depend on the individual engine, mission profile, environment and engine handling.

Another ground station task is the preparation of initialisation data for the on-board part of the system. These initialisation data consist of identification and configuration of the engine (as actually built), and the current state of life consumption for all monitored areas. These data are to be transferred to the on-board system when a new (or repaired) engine has been installed or after the solution of problems with the engine monitoring system (e.g. data corruption or loss).

The database system should be able to provide suitable outputs for maintenance and logistic purposes. Different output formats are required for the different levels of decision making. These are as follows.

- Flight line (First line - at least a GO - NO GO decision must be possible.);
- Engine shop (Second line - life usage data for the complete engine, modules and parts must be available to identify useable parts and the remaining life for each part. For planning of manpower and resources, forecasts of engine removal and the amount of maintenance work are helpful.);
- Depot and overhaul facilities (Third line - detailed life usage data are required, supported be parts history information and reported problems.);
- Engine manufacturer (Fourth line - in addition to the detailed life usage data and parts histories, information about reliability and maintainability problems is important. The aims are improvements in maintenance support, life extension and improved engine design.);
- Logistics (Statistical and individual life usage data are required for deployment planning and stock keeping management.);
- Engineering (Statistical and individual life usage data are required to evaluate existing life limits and design in the event of a failure).

Output is required in the form of formatted reports, data tabulations and plots, and may be displayed on screen or printed. Output from the ground station may also be communicated to on-board parts of the monitoring system or other ground computers using appropriate data transfer media.

A very important point - that should be given special attention - is the 'human' interface of the database system. Not only the output formats but also the complete handling should be user-friendly (i.e. easy, clear, quick, robust). When planning and introducing a monitoring system, one should be aware that acceptance or non-acceptance of the whole system mainly depends on the ease of handling, and on the user-friendliness of the 'human' interface.

### 5.4. DATABASE MAINTENANCE TECHNOLOGY

It is desirable that ground stations be sufficiently flexible to handle communication with different engine monitoring systems. Database architecture and data management logic should be general. Configuration management requirements for database system hardware and software should be established with this in mind. 'Object oriented' concepts which tie the data format closely into the management code are being introduced in modern systems, and assist the construction of such systems.

Database maintenance is necessary to ensure data accuracy and database integrity. Suitable means are trend analyses and plausibility checks, recording of findings and corrective actions as well as regular (preferred daily) backups. Plausibility checks for inputs into the database should be performed in a similar manner as for on-board inputs. One should be aware that errors could occur in all types of data at any point in the data path between original source and the database. Particular attention should be paid to manually entered data, which should be checked to ensure conformance to the underlying data specification. Automatic data-validation checks can be built into the 'business rules' at table level to ensure that a minimum level of consistency is enforced.

The system should be flexible enough to allow for changes of part related data (e.g. life limits) and the introduction of new or modified (improved) design of engine components and parts.

## 6. MISSION CHARACTERISATION

Missions can be characterised in different ways. For engine life-usage, monitoring a description of the engine manoeuvres and their sequential appearance is important. A number of engine manoeuvres can be directly related to aircraft manoeuvres. Indeed, it has been said that pilots fly the aeroplane, and the engine goes where it is taken! Nevertheless, it should be noted that some aircraft manoeuvres do not always require the same engine reaction. In the late 1970s, UK Engine Usage Monitoring System (EUMS) results showed that two pilots flying exactly the same route used very different amounts of engine life. One used the throttle to climb over some hills, and the other used the elevators. Typical manoeuvres which appear in almost every mission include engine start-up, taxi, take-off, climb, cruise, approach and landing, thrust reverse and engine shut-down. Where there is a possibility of different styles or habits developing, they will exist, and will cause life to be consumed at different rates. Combat and aerobatics manoeuvres are also typical

of military use and probably create even greater variations. The most accurate way of allowing for such variations is to fit a life usage monitoring system to every engine. However, this is not always possible and it may be necessary to conduct mission analysis to generate typical mission profiles.

### 6.1. MISSION TREATMENT FOR ANALYTICAL PURPOSES

Mission analysis is done to predict engine life consumption for mission profiles which are composed of the above mentioned manoeuvres. Two differing aims exist:

- To establish the profile of a particular mission and to predict its particular life consumption;
- To create a 'typical' mission profile.

A 'typical' mission profile may be used to predict life consumption in a general way and to derive cyclic exchange rates or other life related information. Improvement of theoretical design missions and refinement of an engine design may also be possible. The basic idea behind the creation of a typical mission is that it should be possible to represent real flight activity in a simplified way, with sufficient accuracy to maintain safety standards.

If this goal cannot be achieved in a single step, the process should be decomposed into smaller parts. The underlying assumption is that it should be easier to relate certain portions of the total life consumption to particular manoeuvres.

The methods used to obtain the required input data are various.

- One method is to equip a single aircraft with a data-recording device and systematically record a number of mission profiles for mission types, which are supposed to represent the typical usage.
- A second method (which is probably much more representative) is to equip a number of aircraft with data recording systems and operate them in the normal way. That is, not selecting them on purpose (but only randomly) for special mission types or deployments.
- A third method (if recording devices are not available) is to ask the pilots. In this case, the pilots are interviewed about type, extent and duration of the manoeuvres they fly, and typical sequences of throttle settings as well as mission types and mission mixes. In the course of these activities a variety of representative missions are established which are simply described by pilot's lever sequences or thrust demand sequences. Historically, pilot interview has not proven to be a particularly accurate method. However, it has provided insight into the qualitative aspects of mission definition.
- A fourth method is via aircraft flight simulators. Simulators are used for training pilots and maintaining their flight proficiency, in lieu of actual aircraft flight time. During this method, the pilots are given a flight syllabus as if they were flying the actual aircraft. Throughout the simulation, data describing the aircraft, engine flight data, and flight conditions can be recorded. This approach is less costly and less accurate than the recording devices aboard actual aircraft but it does produce greater fidelity than simple pilot surveys. It also allows the effects of the expected dynamics of new aircraft and engine types to be explored before they physically exist.

### 6.2. 'BUILDING BLOCK' TECHNIQUES

The technique described here tries to build up a complete mission profile by combining a number of blocks, which contain the profiles of the engine parameters corresponding to certain manoeuvres.

In a first step, a representative number of mission profiles should be analysed to identify both the manoeuvres that occur and the frequency at which they occur in each mission profile. Recordings for all identified manoeuvres should be obtained and separated into blocks. Attempts should then be made to determine the life usage related to particular manoeuvres or blocks, respectively. If the life usage of a special type of manoeuvre is found from this analysis, then an average profile of such a manoeuvre can be created.

Although the blocks are similar in the way that they contain the engine parameters of a certain manoeuvre, they can be very different in the content of the manoeuvre. The types of manoeuvre may range from simple steady-state conditions (e.g. cruise at some constant power setting) to highly complex sequences of throttle movements (e.g. aerobatics).

Having created a typical profile for each manoeuvre, and ascertained its frequency, one can construct a typical whole mission profile. Care should be taken to ensure that the transitions from one manoeuvre to the next are smooth. This task should be much easier than the manipulation of a whole mission profile. Assumptions in this process are:

- That each manoeuvre is independent of those preceding and following it;
- Life usage is dependent only on the manoeuvre under consideration;
- Life usage is largely dependent on the transient elements of engine usage.

### 6.3. TRANSLATION OF MISSION ELEMENTS

In the previous paragraph, it was assumed that recorded profiles were available for missions and manoeuvres. This is not always the case. In many situations, only a description of the aircraft manoeuvres is given and the associated engine parameters must be derived, or deduced.

The general process is that a profile of thrust demand is calculated based on the aircraft performance. This requires the availability of an aircraft performance model, which generally is developed by the aircraft manufacturer. Depending on the type of manoeuvre (e.g. close formation, tanking) a certain pattern of thrust modulation must be superimposed. In addition to the thrust demand

profile, the intake conditions (e.g. described by altitude and airspeed) are requested, and are normally included in the description of the aircraft manoeuvres. Based on a model for engine intake performance, developed by the aircraft manufacturer or established in close co-operation between aircraft and engine manufacturer, the engine intake parameters (in terms of total temperature, total and static pressure) can be deduced.

In the next step, an engine performance model is employed to calculate the profile of the pilot's lever position and the corresponding engine parameters necessary for life usage.

Generally, the engine parameters are required as transient data (i.e. describing the transient response of the engine to modulations of the pilot's lever). This means that the engine performance model must be capable of providing transient outputs. It is well recognised that accurate prediction of transient engine behaviour is much more difficult than the prediction of a steady-state response. Nevertheless, fine-tuning of the performance model should be carried out using recorded engine parameters.

### 6.4. GENERATION OF 'SYNTHETIC' MISSION PROFILES

With the tools and mission blocks described above, one can generate 'synthetic' mission profiles. These mission profiles can be composed as a (nearly) arbitrary sequence of all of the available blocks. Of course, some restrictions exist: every mission should begin with an engine start-up sequence, followed by taxiing (of variable duration) and take-off. In the same sense, every mission should be finished by an engine landing and shutdown sequence. Any other technically necessary limitations in the sequencing of manoeuvres should be observed in order to obtain sensible profiles.

Another requirement for a synthetic mission profile should be that it contains only plausible data. This means that plausibility checks should always be passed. This includes that the engine parameters do not exceed their predefined range and that transitions between subsequent blocks are smooth enough not to violate the predefined change rates.

### 6.5. SUMMARY

This chapter provides details of usage tracking, cyclic exchange rate, data selection criteria, data transfer, records, and accounting and mission characterisation. The need for usage tracking is justified and the different methods are discussed. The method of relating hourly usage rates to 'design cycles' is described in detail. The collection and storage of raw data in bulk form is discussed and recommended. Input parameters are defined in terms of sampling rate, accuracy, repeatability, and the necessary plausibility checks are defined.

Operational and management aspects of usage monitoring systems are described with emphasis on ease of use and delivered value. The tasks of a ground station and its configuration are described. Amongst these is mission analysis, which is described in detail.

## 7. CONCLUSIONS AND RECOMMENDATIONS

- Before introducing life-monitoring systems, all organisations should identify the problems that they wish to solve, and the benefits that they wish to obtain.
- To maximise the benefits, the user must thoroughly prepare for a change in working practices before a modern life usage monitoring system is brought into service.
- Priority must be given to the relevance of information available at the ground station, and robustness and simplicity of use.
- Because fatigue damage of a critical area is related to its individual stress-temperature history, calculating life consumption for individual engines or critical components is the best way to avoid the premature removal of components.
- Pilot survey is the least expensive and least accurate method of developing mission definitions. Installation of in-flight data recorders in a random sample of the fleet aircraft to capture actual flight data is more costly, but is the most accurate method.

## 8. REFERENCES

SAE AIR 1872 - Guide to life usage monitoring and parts management for aircraft gas turbine engines

SAE ARP 1587 - Aircraft gas turbine engine monitoring system guide

SAE AIR 4061 - Guidelines for integration of engine monitoring functions with on-board aircraft systems

SAE AIR 4175 - A guide to the development of a ground station for engine condition monitoring to scan.

# Chapter 9

## Usage Data from Operational Monitoring Systems

by

*(M. Sapsard)*

# 1. INTRODUCTION

Once all of the research and development described elsewhere is finished, an aircraft, engine, and engine monitoring system will arrive in service. This equipment will be used in accordance with the manufacturers instructions and any flight safety requirements imposed by the airworthiness authority. Whatever debate there may have been earlier, and whatever scientific and statistical uncertainty there may be about the following:

- The materials;
- The algorithms used;
- The implementation;
- Cross correlation with the manufacturers design model;

the life usage counts recorded by the life usage monitors will provide 'REAL' life usage counts to the operator and the authorities. These numbers will be accepted 'warts and all', and are the numbers that we discuss in this chapter.

In this chapter we discuss and show the following.

- The statistical basis for deciding how many aircraft, engines or stress features to monitor.
- Some real data collected from in-service aircraft engines.
- How the data has been interpreted.

# 2. FIRST CONSIDERATIONS

When a working monitoring system is to be fitted to an aircraft there are a number of questions that usually have to be answered. The following are probably the most frequently asked.

- Which features should be monitored?
- How many engines on how many aircraft should be monitored?
- How can we use the results?
- How will we know what we are looking at?
- Should we continue to measure life in flying hours, or change to using life usage counts as recorded by the monitoring system?

The simple answers to these questions are:

- The first and most simple rule in an operational environment, rather than a research environment, is 'Don't collect data unless you already know the decisions you intend to make with it'. So if you do not already know what you are looking for and why, then do not do it on operational aircraft. The only exception may be when a trial is being deliberately conducted.
- This depends on the sample size and the variation within it. Unfortunately, in statistical terms the size of the fleet is a secondary factor in determining how many aircraft or engines should carry monitoring equipment.
- This depends on the relationship between the operator and the manufacturer, the fleet size, and the technical capabilities of the operator.
- In a deeply technical subject area, specialist help may be needed.
- This question has yet to be answered by most operators and manufacturers. One practice is for the authorities and the operator to work in flying hours, and for the results of the monitoring system to be converted from 'Counts per flight' to 'Counts per flying hour'. This is known as the 'Beta' factor by several European airforces and manufacturers. Once known different Beta factors are often applied to components by airbase, engine position, engine or aircraft modification state, or any other significant factor that is discovered.

# 3. THE STATISTICAL ASPECTS OF MONITORING

When data is collected it can only have any statistical validity if there is enough of it, and if tests on the data show that it conforms well enough to one of the commonly known distributions for the results of any further mathematical analysis to be valid. Even if the raw data does not conform well, it may be possible to transform it into a distribution that does. An example of this is the log-normal curve.

The following shows, in detail, how to calculate the answer for lifed components when their usage distribution approximates closely to a normal distribution. A graph that permits informed decisions to be made about sample size is also derived. The worked example is followed by graphs for lifed features which have a usage distribution that approximates to the normal distribution.

Graphical displays allow non-statisticians to obtain an insight into what is going on. However, it must be remembered that they are only illustrations and that a mathematical analysis and proof of conformity and hence validity is essential before statistical analysis is performed.

## 3.1. HOW MANY ENGINES IN A FLEET SHOULD BE MONITORED?

It is sometimes assumed that if a fleet of several hundred engines is in use then monitoring, for safety purposes at least, should be adequate if a small number are monitored. It is well known that on production lines for mass-

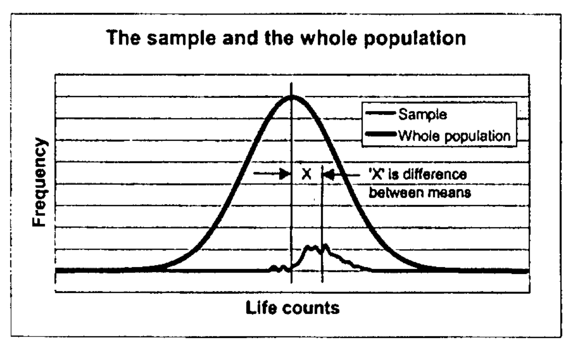



Figure 1 – Population and sample relationship

produced car components, maybe only one in a hundred components is measured. This is enough to ensure that very exacting standards are met. Unfortunately there is a vast difference between the two situations. On the

production line most of the factors that cause variations have been identified and brought under tight control. Whereas, the military aircraft, and more particularly, the engine usage spectrum has not.

Let us assume that the measured parameters of each of the above populations fit a standard distribution curve. Although the number of samples for each may be similar, one is likely to have a very small variance or spread while the other has a large one. The following analysis is from Sapsard MJ, Flanigan A, Pitwood IR (1996).

Because the sample only covers part of the population there is some uncertainty about whether the average of the sample is the same as that for the whole population. Figure 1 shows this difference as 'X'.

In safety critical situations some allowance must be made for the possibility that the mean of the sample is less than the true whole population value. However, if the fleet is large enough, as the sample size increases the smaller the uncertainty about the difference between the means will become.

When the sample variance and size are considered for a fleet or population of various sizes we can construct the

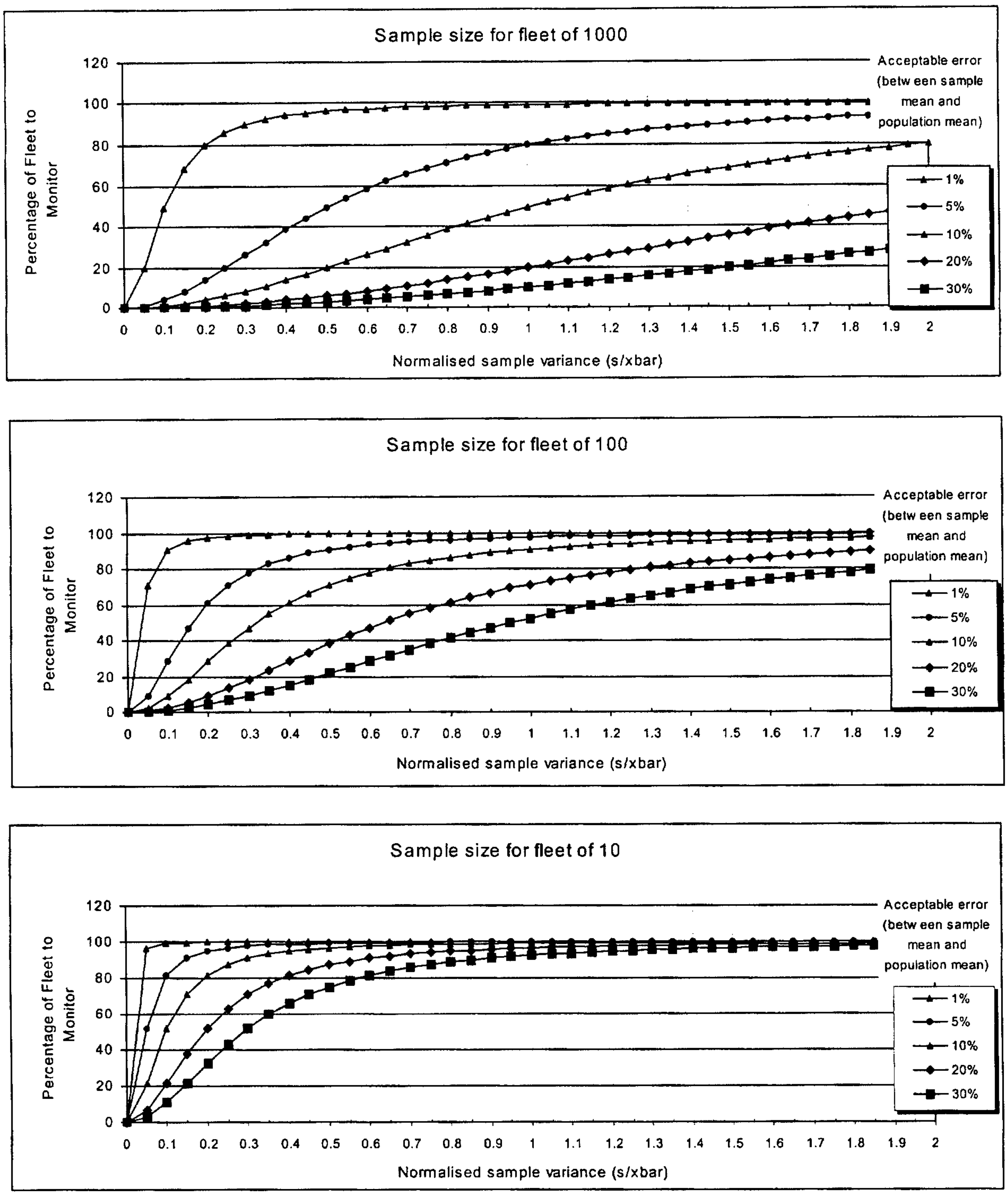



Figure 2 – Sample size dependency on sample variance, for fleets of 1000, 100, and 10 engines

curves in figure 2. These show what has to be done if the operator wishes to ensure that the size of 'X' shown above is less than say 10% of the value of the mean.

If the shape of the sample is close to the normal distribution, it may be assumed that the whole population has a similar distribution. We also know the whole population size - N. In this chart $\mu$ is the mean of the whole population. From the sample, we can estimate the range within which $\mu$ falls, by using Student's t. This states that:

$$\mu = \bar{x} \pm \frac{ts}{\sqrt{n}}$$

where t is a measure of the confidence level (a probability between 0% and 100%) that can be given to the new estimate of $\mu$, and gives an estimate of the range of values within which $\mu$ must fall. A confidence level of 99% would necessitate a larger value of t and hence a wider range of values within which $\mu$ could fall. This would mean that there was only a 1% chance of $\mu$ lying outside of the estimated range of probable values.

By considering the sampling requirements for a finite population it can be shown that the sample size for a given value of the difference between the means, as a proportion p of the mean of the sample is given by:

$$n = \frac{N}{(N-1)\left(\frac{p\bar{x}}{ts}\right)^2 + 1}$$

By substitution, the graphs in figure 2 may be obtained for fleets of 10, 100, and 1000 components.

The following assertions should be noted.

- The number to be sampled depends primarily on the variance (or degree of scatter) of the sampled data and the sample size, rather than on the fleet size. This means that the sample size required for a statistically valid sample could be larger than the size of the fleet.
- Once the desired sample size for a particular fleet or aircraft type has been determined, a doubling of fleet size, with no change in the way in which the aircraft are used, requires a 41% increase in the number of sampled aircraft.
- The charts derived here can, after appropriate tests for applicability, be applied to lifed components in any mechanical or other system.
- The method can be modified and applied to any lifed component with a usage distribution that can be mapped onto a normal distribution. An example is the log normal distribution.
- At least 30 samples should exist for each statistical degree of freedom, for each permutation of variables such as squadron, operating base, sortie pattern, life-usage affecting modification, time of year, etc.

## 3.2. USING THE CHARTS

To decide how many engines in a fleet should be monitored it is necessary to have enough sample data so that the normalised sample variance can be determined in a statistically valid way. For guidance, typical values observed on the UK Harrier and Tornado aircraft lie between about 0.4 and 1.6 depending on the selected components. Most components have values between 0.4 and 0.7.

As an example, let us assume the following:

- We have a fleet of 100 aircraft;
- The variance is 0.5;
- We will accept an error between the mean of the sample and the mean of the fleet of 10%.

From the middle chart we can see that about 72% of the fleet should be monitored.

# 4. MEASURED RESULTS FROM OPERATIONAL AIRCRAFT

## 4.1. RAF HARRIER

When the Harrier GR Mk 5 entered service in 1989 a sample of Pegasus Mk 105 engine-component usage-data from about the first 1700 flights of the first twelve aircraft was collected. This data was collected at the rate of eight samples per second with real time on-board computation. The fleetwide values of $\mu$ for all components are shown, in table 2, for confidence levels of 99%, 95% and 90% that $\mu$ will lie inside the estimated range.

We now have a measure of the likelihood that $\mu$ lies within the range indicated. In the worst case (largest range), which is also the safest assumption in terms of airworthiness, the 99% confidence level should be adopted, and $\mu$ = 5.17+/-2.82 as shown in table 1.

When the above curves were applied to the variances obtained from the Harrier GR Mk 5 Pegasus engine table 2 was obtained.

| Confidence Level % | t | $\bar{x}$ | Range of $\mu$ |
|---|---|---|---|
| 99 | 3.1 | 5.17 | 5.17+/-2.82 |
| 95 | 2.2 | 5.17 | 5.17+/-1.99 |
| 90 | 1.6 | 5.17 | 5.17+/-1.63 |

Table 1 – Range within which the fleet mean should lie

This shows that for the high pressure turbine at least 72% of the fleet should be monitored if an error between the sample and population means of less than 10% is to be achieved. In the case of this engine it was decided to fit a fleetwide engine monitoring system before it entered service.

### 4.1.1. VISUALISATION OF RESULTS

The figures 3 and 4 show what some of the Pegasus usage distributions look like. The first is of high-pressure compressor life usage across several engines. It can be seen that two distinct distributions appear to exist. This is because two squadrons were operated in the first year of

service. One was a training squadron and the other an operational unit. The dual (twin peaked) distribution is because the aircraft were not shared between the squadrons, and the usage was different. This was clearly shown by the detailed statistical analysis.

| Difference between sample and population means | 0% | 5% | 10% | 20% | 30% |
|---|---|---|---|---|---|
| Component | % of Fleet to be monitored | % of Fleet to be monitored | % of Fleet to be monitored | % of Fleet to be monitored | % of Fleet to be monitored |
| LPC | 100% | 93% | 78% | 61% | 47% |
| LPS | 100% | 97% | 89% | 79% | 68% |
| LPT | 100% | 96% | 86% | 73% | 61% |
| HPC | 100% | 92% | 74% | 56% | 42% |
| HPT | 100% | 91% | 72% | 54% | 39% |
| CCOC | 100% | 97% | 88% | 76% | 65% |
| THF | 100% | 94% | 80% | 64% | 50% |
| Creep | N/A | N/A | N/A | N/A | N/A |

Table 2 – RAF Pegasus engine: percentage of fleet to be monitored to limit the differences between the sample and fleet means to specified limits.

The skew (long tail) to the right appears to be typical of measured life usage distributions for fatigue limited components. Typically, results for the Pegasus engine in the Harrier aircraft have revealed approximately Poisson distributions for individual engines and Normal distributions for the whole fleet. From a general statistical

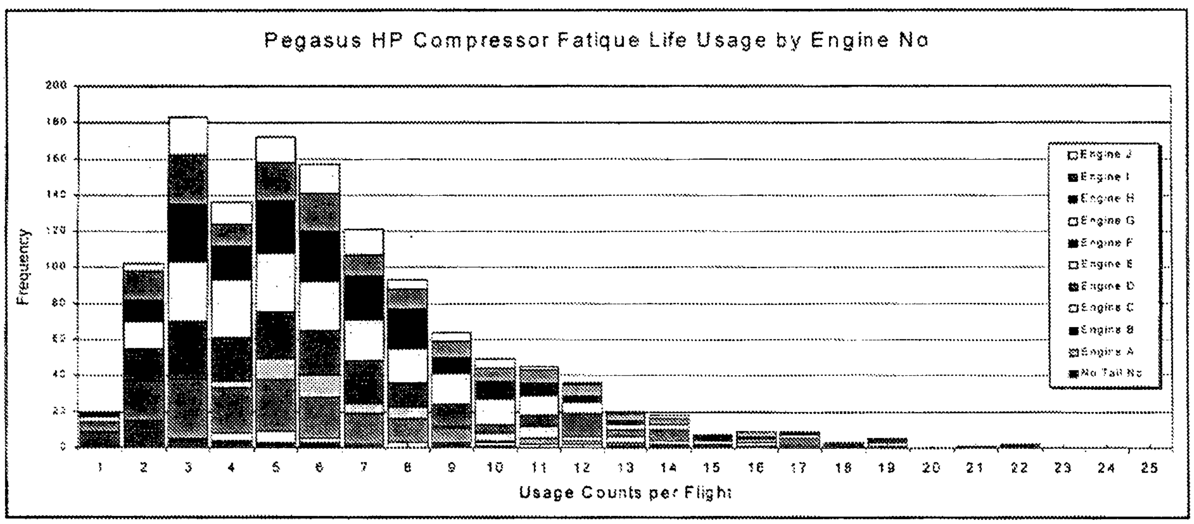



Figure 3 – Pegasus hp compressor life counts

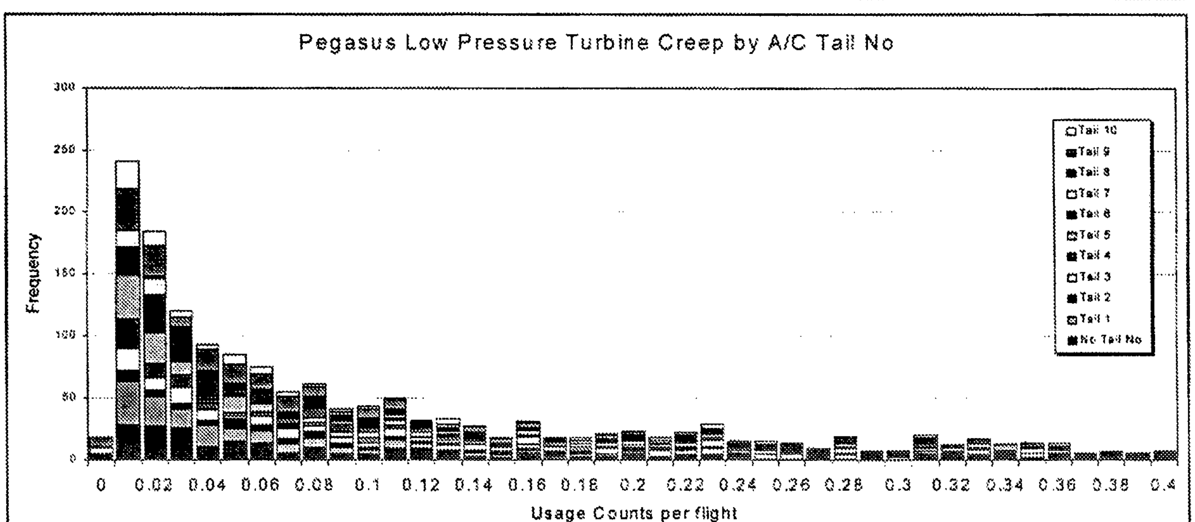



Figure 4 – Pegasus lp turbine blade creep life counts

viewpoint the variation in observed life usage counts is high. This is because the standard deviation is at least 50% of the mean throughout. To put this into context this ratio is usually in the order of 20% to 30%. It was also found that better correlation was obtained if the data was converted from 'counts per flight' to 'counts per hour'.

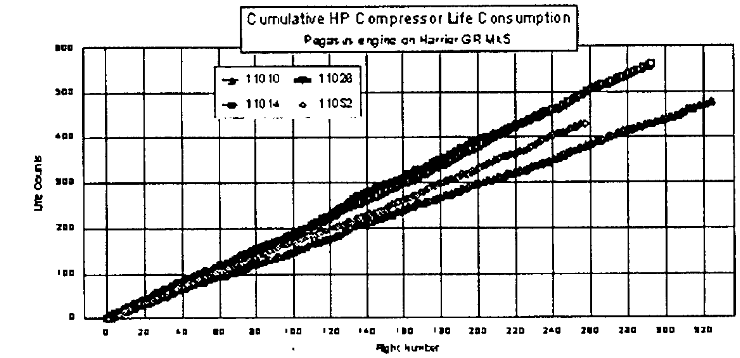



Figure 5 – Variation between engines, in cumulative life consumption

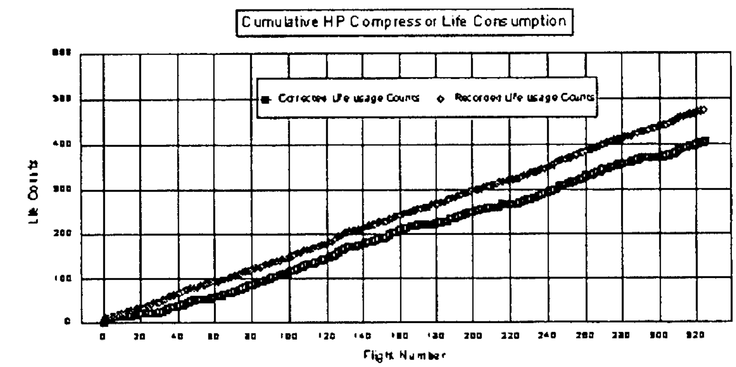



Figure 6 – The effect of replacing lost data

Figure 4 shows low-pressure turbine blade creep usage distribution across the same group of aircraft. This is not strictly relevant to critical parts usage, but illustrates several points. Clearly different rules apply here, and the normal distribution cannot be used unless this distribution could in some way be mapped onto the normal curve.

In this case it is worth considering how the Harrier aircraft is operated, and also looking at the cumulative usage curves for individual engines, for a possible explanation of this usage pattern.

The Harrier aircraft has a hovering capability, which dramatically shortens turbine blade life if used extensively. All pilots must practice flying in the hover to maintain their skills. This manoeuvre is also demonstrated at air displays when the aircraft 'bows' or 'nods' to the audience. These activities help to produce the long tail to the right. Creep is essentially a time and temperature phenomenon. Because the turbines are designed for acceptable life at the hot end of the operational envelope, the cooling efficiency at normal operating conditions is very high and very little creep damage occurs.

Figure 4 shows how creep life usage can have very high values. These very high values are all associated with air show demonstrations.

### 4.1.2. Replacing Missing Data

In figures 3 and 4, one aircraft is described as 'No Tail No'. This is because of inadequate labelling of collected data. Although the data could not be applied to any particular component, it was not discarded because it added to the total knowledge about the fleet of components.

Figure 5 shows the cumulative life consumption for the hp compressor for four different engines. Others have been removed for clarity. It can be seen that two lie very closely together, but that the other two are quite separate. This form of presentation is very useful for two reasons.

- It removes the confusing effect of scatter (This will be apparent from figures 7 and 8);
- If two components show a distinctly separate line the difference is probably statistically significant.

The difference in the rate of life consumption between the best and worst engines is about 20%. If this information is made available to the fleet manager, and included in fleet planning considerations, it should be possible to achieve a number of advantages. These include the following.

- Identification and rectification of engines with badly matched, or inefficient components.
- Identification and rectification of aircraft with poor intake efficiency.
- Identification of pilots, who consume more or less life than others. This may be used to advantage by their trainers.
- Better selection of aircraft for tasks, which are known to consume life more rapidly.
- Overall fleet life management.

In figure 6 the lower line shows the cumulative recorded hp compressor life usage on a particular aircraft. The horizontal parts of the line show where no data was collected.

The upper line shows the effect of replacing the missing data with the average usage for that component on that aircraft over the entire period. The effect is to account for approximately 17% additional usage. The point of the observation is that although the visual difference between the lines may not appear to be very great, the real difference in safety and operational terms could be a burst disk.

## 4.2. GAF TORNADO

The Tornado aircraft is operated by four nations: Germany, Great Britain, Italy and Saudi Arabia. Each operating environment is different and engines for each differ in detail such as cooling hole design in the turbine blades. These and other differences lead to differences in the life usage rates between each group of engines. The GAF fleet is equipped with the On-board Life Usage Monitoring System (OLMOS) engine monitoring system, which has provided engine life usage data from many thousands of flights, J Broede and H Pfoertner, (1997).

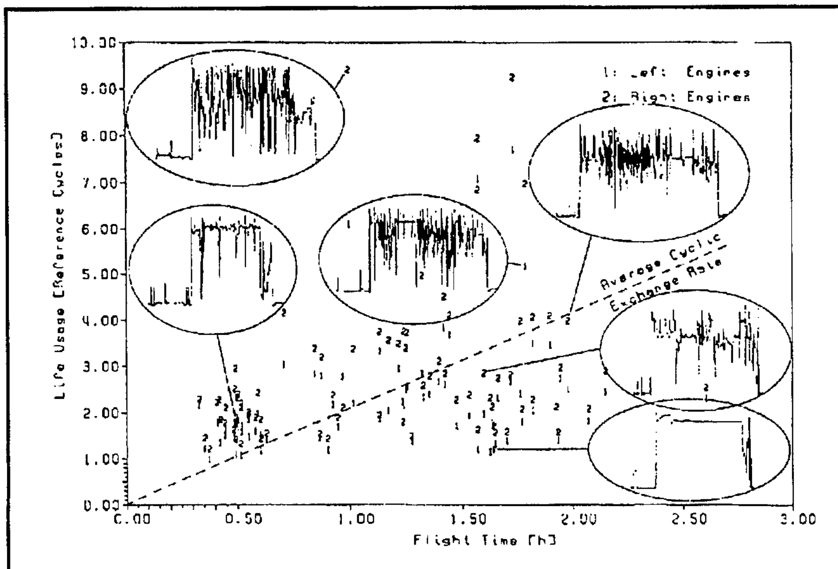



Figure 7 – Usage variations from GAF Tornado fleet

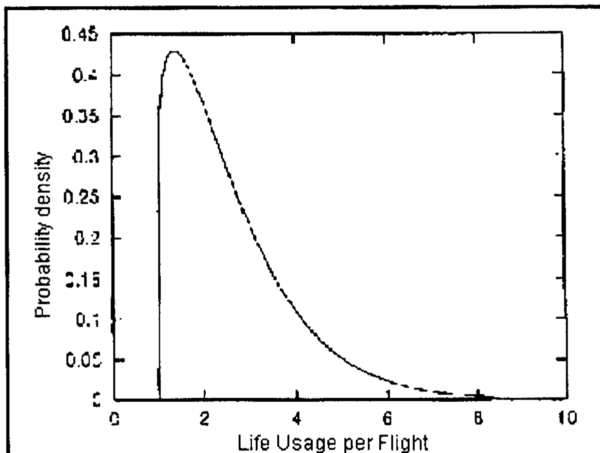



Figure 8 – Statistical distribution of life usage

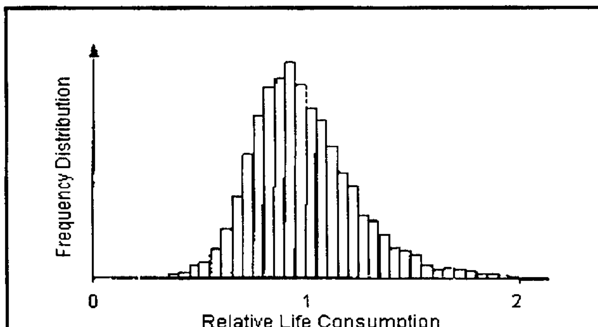



Figure 9 – Relative life consumption – RB 199

### 4.2.1. VISUALISATION OF RESULTS

The overall results confirm the RAF Harrier experience. Figure 7 indicates the scatter found in life usage for the intermediate pressure compressor over 77 flights.

From this, the following have been deduced.

- Clusters of similar usage values appear, and may be associated with a particular mission.
- Significant differences in usage exist between the right and left engines, due to engine handling procedures (namely, which engine is first switched on and off, and the resulting warming up and cooling down periods).
- The correlation between flight time and life usage is not as strong as might be expected.
- There is a sharp cut-off for low life usage values, that is independent of flight time, due to the minimum power requirement for take-off, and to the start-up and shut-down stresses.
- Differences in engine response to the control system and differences in settings (resulting possibly in overshoots at the end of accelerations) have also been shown to contribute.

### 4.2.2. STATISTICAL MODELLING

Figure 8 shows the result of fitting a 3-parameter Weibull distribution to the life usage data from figure 7. To do this the flight time information has been discarded. Therefore, the distribution function describes the difference between engines, but not different flights. Scatter in life consumption between different flights has already been smoothed due to the accumulating nature of life consumption. It was noted by the authors, J Broede and H Pfoertner, that the small number of high life-usage points were not very well approximated by this distribution, and that this is one of the reasons for individual engine monitoring. If this is compared with the distribution for the Harrier shown in figure 3 there is sufficient similarity to suggest that these findings may be generally true.

The only thing that may change this is an improvement in the underlying algorithms, which brought new factors into consideration. However, to do this would cast severe doubts on the existing airworthiness of the components concerned. Because air-worthiness is subject to formal procedures and is therefore (hopefully) well proven, it seems likely that future changes will be refinements of the existing procedures.

Two examples of such improvements exist.

- Within the GAF a thrust-rating system is used for take-off. This has resulted in the observation of an

increase in life-usage rates against time in service, and consideration is now being given to including the deterioration of seals and cooling air flows in future life usage models.

- In some engines the transient thermal stresses are now included in the dynamic models and much more accurately modelled than in the past.

#### 4.2.3. FLEETWIDE LIFE USAGE

The fleetwide normalised probability distribution function for life usage across all RB 199 fracture critical components is shown in figure 9. This is a Normal/Poisson distribution, with some skew to the higher values. There is a considerable smoothing effect due to the varying life usage characteristics, but the lower cut-off point of about 0.5 is clear.

Figure 10 may be of more use to fleet managers. It shows how the components in the fleet are consuming life.

Figure 10 shows the distribution of life usage in the fleet for an intermediate-pressure compressor disk. The following observations have been made.

- The left-hand tail shows spares and components that have recently been built into engines.
- The majority of parts are those which were built into new engines - between the 30% and 80% life usage bands.
- The right-hand tail shows components that are soon to be withdrawn from service.

The high-end components are of greatest significance for safety and planning purposes. They represent fast/hot running engines, and the repeated assignment of certain aircraft to particularly demanding missions. Similar charts can be derived for all components. This is often a task for the fleet management systems, and is discussed further in chapter 8.

### 4.3. FAF MIRAGE 2000

The SNECMA M53P2 engine powers all Mirage 2000 aircraft. The Engine Life Monitoring System (ELMS) calculates and tracks the creep and fatigue damage for about 20 critical locations. The life history is recorded and tracked by component and engine module. If components are moved to another engine, the individual component data is also transferred, and stored in the ELMS. The measured usage is therefore independent of engine configuration, and characteristic of the squadron missions.

The current operational usage unit is the Engine Flying Hours (EFH). The damage related unit is the Mixed Mission Unit (MMU), which represents the damage caused by a weighted mix of different flight profiles. Initially, this was thought to represent the mean flight profile. However, it became apparent that usage scatter was important, and that no aircraft actually flew the mean flight profile. The ratio MMU/EFH is a measure of the severity of the flight profile, and increases with throttle excursions and turbine time-at-temperature.

The data in figures 11 to 14 was collected from two bases, each with a different operational mission, over half yearly and annual periods from 1992 to 1997. For the example,

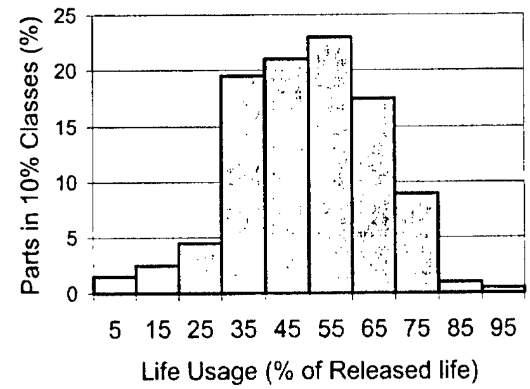



Figure 10 – Life usage distribution for fleet manager

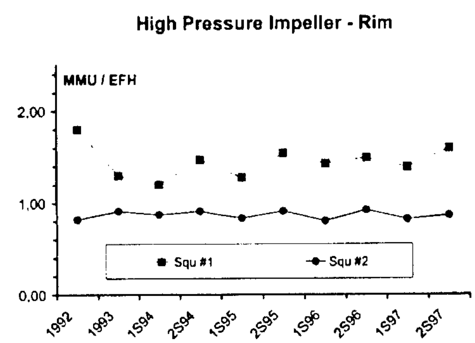



Figure 11 – HP compressor mean usage by airbase

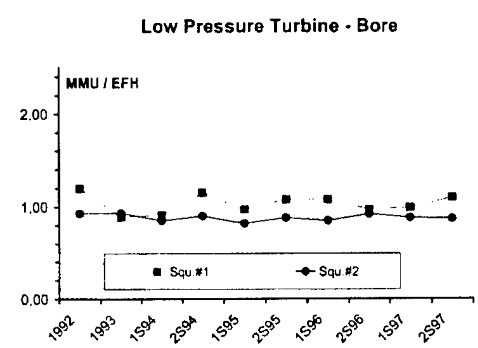



Figure 12 – LP turbine bore mean usage by airbase

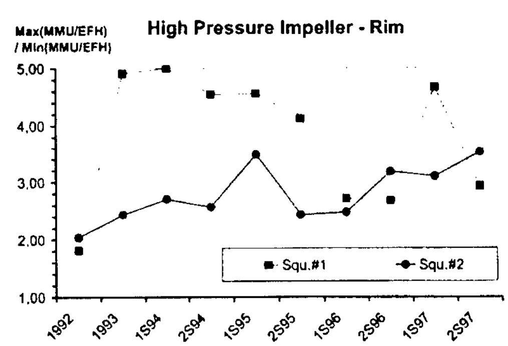



Figure 13 – HP compressor scatter by mission type

two critical features have been examined. These are the disk rim at the high-pressure compressor exit, and the bore of the low-pressure turbine. The first is subject to creep and fatigue, and the second to low cycle fatigue at low to medium temperatures. Figures 11 and 12 show the long-term trend in the ratio MMU/EFH for each base. It can be seen that the mean usage rate is relatively constant for both bases over the period. The differences are predictable: one mission is a high-altitude fighter mission, and the other is a low-altitude navigation based mission.

The range of usage scatter for each mission type, over the same intervals, is shown in figures 13 and 14 as the ratio of maximum MMU/EFH to minimum MMU/EFH. Here it can be seen that the scatter varies considerably along the time-line, from about 1.5 to 5. This result was unexpected, and meant that the mean and range of usage were independent of each other, for this fleet. This means that monitoring the fleet mean is not sufficient to ensure that safety standards are maintained. Care must be taken to ensure that the worst component remains in a safe condition whatever this ratio may currently be.

A possible shortcoming of this technique is that the ratio of maximum MMU/EFH to minimum MMU/EFH is more sensitive to a 1% change in the minimum value than it is to a 1% change in the maximum value. Although this should be considered when using these diagrams, the underlying message is clear.

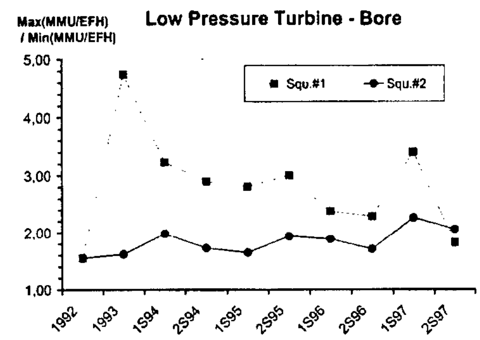



Figure 14 – LP turbine bore scatter by mission type

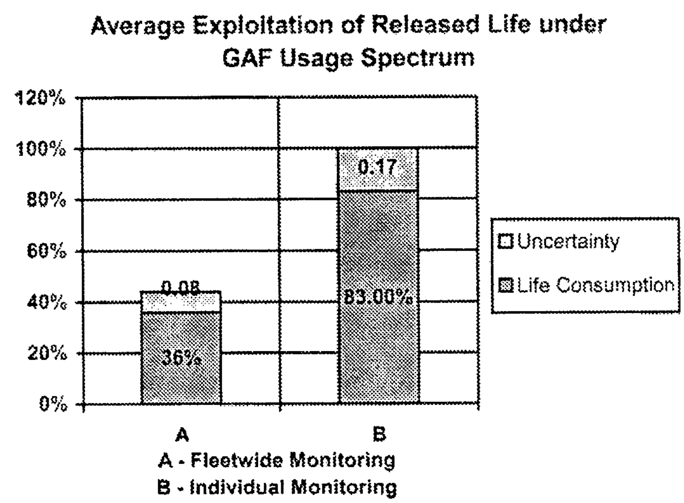



Figure 15 – Fleetwide versus individual monitoring

## 5. INDIVIDUAL ENGINE MONITORING

It has been shown elsewhere, Broede J (1995), that individual monitoring of component life-usage can more than double the available life for a given part. There must always be a point beyond which the marginal cost of gaining another cycle from a part is greater than the benefit. However, it has been shown that the replacement cost may exceed the purchase price of a part by a factor of 10. Figure 15 indicates the additional life that may be attainable with a life monitoring system. This is based on GAF experience with the RB199 engine. The uncertainty region is a measure of the known inaccuracies of the algorithms used. Clearly much more life may be extracted with individual engine monitoring, than with fleet sampling techniques. In this example, a factor of two is suggested.

## 6. SUMMARY

The following have been shown.

- How to estimate the number of engines to sample, in a fleet.
- The number of engines to be monitored primarily depends on the sample size and sample variance, rather than on the size of the fleet.
- The usage variation is typically so high that most of the fleet should be monitored to obtain a statistically valid estimate of usage. Operational convenience may then lead to a decision to monitor all engines.
- There are statistically significant variances between usage rates on nominally 'identical' engines in 'identical' aircraft.
- The variance or range of scatter varies significantly with time.
- That missing or lost data should, at least, have average values substituted in their place.
- Individual engine monitoring potentially offers twice as much usable component life as fleet sampling.

## 7. REFERENCES

Sapsard MJ, Flannigan A, Pitwood IR, *'Analysis of Usage Data from the Pegasus Mk105 in the Harrier GR Mk 5"*, Aircraft Health and Usage Monitoring Systems - Selected Papers AeroTech 95, IMechE Seminar Publication 1996.

J Broede and H Pfoertner '*OLMOS in GAF MRCA Tornado - 10 Years of Experience with On-Board Usage Monitoring*' 33rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, July 6-9, 1997 / Seattle, WA

Broede, J., '*Design and Service Experience of Engine Life Usage Monitoring Systems*', 5th European Propulsion Forum, Pisa, 1995.

# Chapter 10
## Conclusions and Recommendations

## 1. INTRODUCTION

Critical components, for which safe operating lives are defined, include the major rotating parts, the turbo-machinery disks and shafts, and structural casings subjected to high loads. In addition to the major components, other components in the rotating assemblies, such as spacers, cover-plates and seals, may also be designated as critical. To avoid unacceptable risk of catastrophic failure it is necessary to monitor the life usage of critical components and retire them from service before their allocated life has been exceeded.

This report examines the methods that are currently in use to manage the operation of life-limited parts in gas turbine engines. The purpose is to make recommendations on the methods which engine operators can use to minimise the costs of this process. The report concentrates on the critical parts (disks and shafts) and covers the needs and the various options for the retrospective application of systems to monitor and manage the accumulation of component usage. It considers engines designed in the period 1950 to 1990, where these systems were not applied at the design stage.

## 2. HIGHLIGHTS

There are different practices in different nations. The discussions provided a forum that increased the understanding of all concerned, about the concepts and practice elsewhere.

Because of the variability inherent in the operation of most military aircraft, the maximum benefit in component life comes from fitting monitoring devices to all aircraft, preferably during manufacture. However, these benefits must be balanced against the costs of producing and installing the units, as well as data management activities.

The relative merits of the 'safe life' and 'damage tolerance' approaches are discussed and explained.

The relative payback of fleetwide monitoring, versus no monitoring was estimated to be about 100% in realisation of potential life.

## 3. SUMMARY AND RECOMMENDATIONS

Several different approaches to the management of life consumption are described and their relative merits evaluated.

The fundamental 'safe life' method employed with engines designed during the period under consideration was based on the standard measure of engine use, namely flying hours. The procedure consisted of calculating, for each component, the minimum life capability and applying a weighted average rate of damage accumulation per flying hour. A life in hours, based on these two values, was then released.

There are several improvements on this basic process which aim to refine the damage accumulation rate (exchange rate) used. These may use different damage accumulation rates for different sortie patterns or different operating environments; or they may use recordings from a sample of operational sorties to estimate a safe rate.

The 'damage tolerance' design approach can be used to achieve extended service lives of critical components, typically beyond the minimum calculated life. The risk is controlled by a sophisticated inspection programme. However, it is not economical to apply this technique for the whole component life because of the costs associated with dismantling engines and carrying out the necessary inspections.

The operation of most military aircraft types is highly variable. There is variability between mission types, and variability between the ways that different pilots fly the same mission. Therefore, the only way to achieve the maximum life from the life-limited parts is to individually track the life of each part. This necessitates the use of monitoring systems.

The main obstacles to retrofitting monitoring systems to existing aircraft are the cost and complexity of the aircraft changes, and the additional costs of the data collection and management activities. These major cost elements have to be balanced against the cost savings from increased component lives.

The best results will be obtained if the engine manufacturer is involved in the production of the system. He must be involved in defining the algorithms used because he has the best knowledge of the engine and the manner in which it responds to variations in usage.

Many engine designs in the 1960's and 1970's did not use computer based finite element stress analysis that included transient thermal effects. These designs should be reanalysed using modern techniques, if the maximum benefits are to be realised. However, there are still significant gains to be made by using the original analysis results with a full-fleet fit of monitoring devices.

The usage monitoring of critical components has a clear relationship with aircraft safety. Therefore the validation process needs to be carefully thought out and the system should have an appropriate level of duplication. This should include, as a minimum, a ground system which maintains a duplicate record of accumulated damage and which is able to provide a critical examination of damage accumulation rates.

Configuration control is a vital element in any usage monitoring system. This applies not only to the monitoring system itself, but also to the engines being monitored and the data that comes from them.

As soon as the control of the life of critical components is separated from the lifing of the total engine (overhaul life), there is a significant increase in the individual component critical data that must be managed. This needs a suitable, high integrity, data handling process covering the transfer of data to and from each aircraft, and data management at all of the operating bases and at the logistic-control centre.

# Appendix 1

## US Military Engine Tracking and Operational Usage Methods

by

*(P. Maletta – G.E. Aircraft Engines)*

Several US military engine applications incorporate on-board engine monitoring systems to provide tracking data which is critical to the successful Life Management of engine hardware in the field. These tracking systems can record various types of cycles based on the size of engine power excursions. Since many items (e.g. controls, electronic components, etc.) relate to time rather than cycles, the monitoring systems can also track engine operating time (EOT) and engine flight time (EFH). For those components that are limited by time at high temperature, dwell time at high temperature levels and time at military (IRP) or augmentor power conditions are tracked.

Engine maintenance plans should be based on a thorough understanding of the engine application's operational usage, and the appropriate limits set for each tracking parameter based on analysis of this usage. An aircraft mounted flight loads recorder (FLR) provides the most complete description of operational usage. Analysis of this data can identify variations in base-to-base usage as well as fleet-to-fleet.

The current system of tracking engine cyclic usage in the USAF fleet is, for the most part, based on the use of Total Accumulated Cycles (TACs). Total Accumulated Cycles are simple gate based counters used to track individual engine cyclic accumulation. They represent a somewhat arbitrary unit of cyclic damage, which relates partial cyclic accumulation to the damage associated with a start/stop cycle.

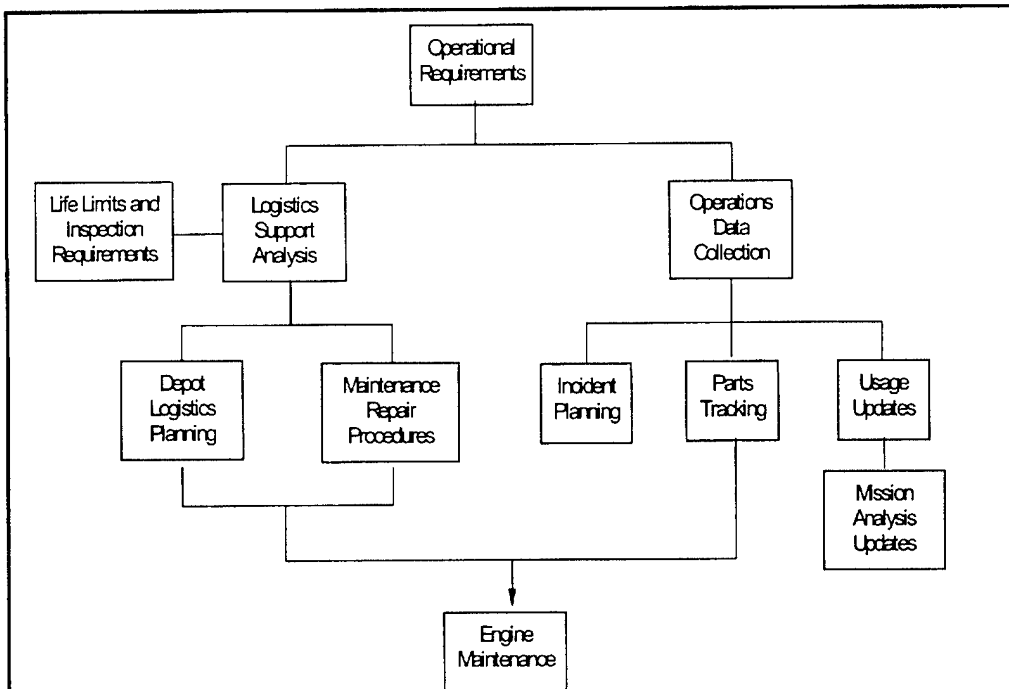


Figure 1 - Operations Data Collection and Mission Analysis Updates

The following equation defines TACs:

$$TAC = LCF + FTC/4 + CIC/40$$

Where:
LCF = Engine Start to IRP to Engine Stop Excursion
FTC = Idle to IRP to Idle Excursion
CIC = Cruise to IRP to Cruise Excursion
IRP = Intermediate Rated Power

Note that "cruise" is not rigorously defined and the implied relationship of damage between the various types of cycles is not necessarily representative of any location within the engine.

Since TAC accumulation is a strong function of the engine parameter monitored as well as the gates and methods that distinguish the different types of cycles encountered, they are not universally equivalent. One application's life limit quoted in TACs cannot be compared with another's unless both count using the same methodology with the same gates. Therefore, an engine's TAC accumulation rate is affected by both the missions flown and the counting system employed for tracking. Furthermore, every counting system, except actual on-board damage systems, must calibrate the life limits used in the parts life tracking system to actual mission severity.

For a given fleet of aircraft and engines, the use of TACs provides a consistent measurement of tracking engine cyclic accumulation. It is through detailed and continual mission assessment that these TAC accumulations are calibrated to a specific engine application's life capability. The timely analysis of engine field usage data and the subsequent determination of the resultant engine damage expressed in TACs is central to the USAF "closed loop life" philosophy. It assures that service limits and maintenance actions are based on the actual field usage of the engine fleet.

The USAF accumulates data on fleet usage through a formalised system that transfers recorded flight-loads data from the field to the engine contractor. Periodic fleet surveys are also conducted to obtain information from pilots. This is used to complement the electronic data. The electronic data is processed through the engine contractor's flight data analysis system on a continuous basis. A fully integrated database allows for the rapid evaluation of mission statistics and the comparison with the previous usage that was used to set the current life limits. A database usage report can provide

extensive information on aircraft and engine usage, which is valuable to many end users.

The mission generation process produces a set of realistic missions that best represent fleet usage. The goal of this effort is to develop a complete definition of engine usage that can then serve as the basis for setting life limits. To attain this goal, representative mission profiles have to be defined in terms of power excursions, altitude and mach number descriptions and combined with the appropriate mission and ambient temperature mixes. Engine deterioration characteristics and transient descriptions are also considered.

Field usage surveys are required to develop an understanding of aircraft and engine usage. This can then be used during the analysis of the electronic flight data. Pilot interviews are very useful in understanding "typical" flight scenarios as well as the types of variations that can be expected in the electronic data. The field usage survey also provides the opportunity to obtain information on ground runs and testing as well as data that may only be available at the bases.

The basic mission generation process, shown in figure 3, is heavily dependent on the availability of Electronic Flight Data (EFD) to develop missions that characterise the fleet's field usage. This electronic data is processed, plotted and analysed by the flight-data analysis software. The resulting summary statistics for each sortie are loaded into a database. Statistics collected include those needed for mission development as well as those of interest to other groups. After all of the flights have been analysed and the appropriate statistics have been loaded into the database, the statistics are extracted for each of the sortie types that are to be modelled. Generally, statistics on pre and post flight time, EFH, EOT, IRP and A/B Time at Takeoff, Total Time at IRP and A/B, Number of A/B Lights, and a variety of cyclic statistics are obtained. The averages, distribution shapes and outliers are then evaluated.

Once appropriate 'target values' are determined, the database is then queried for possible candidate missions that best match the cyclic content and time at power statistics. The candidate missions are reviewed to determine which one profile best demonstrates the characteristics that the pilots described as being most typical of that sortie type. The 'Best Fit' candidates are the starting points from which the final field missions are derived. Damage studies show that the amount of damage can be affected by where in the flight envelope engine cycles occur, and that the cause of stress varies for each location throughout the engine. Hence, the goal of the mission generation process is to match all of the statistics for each mission category or sortie type. Matching statistics by mission category helps assure that the resultant model mission is representative of each category's fleet utilisation.

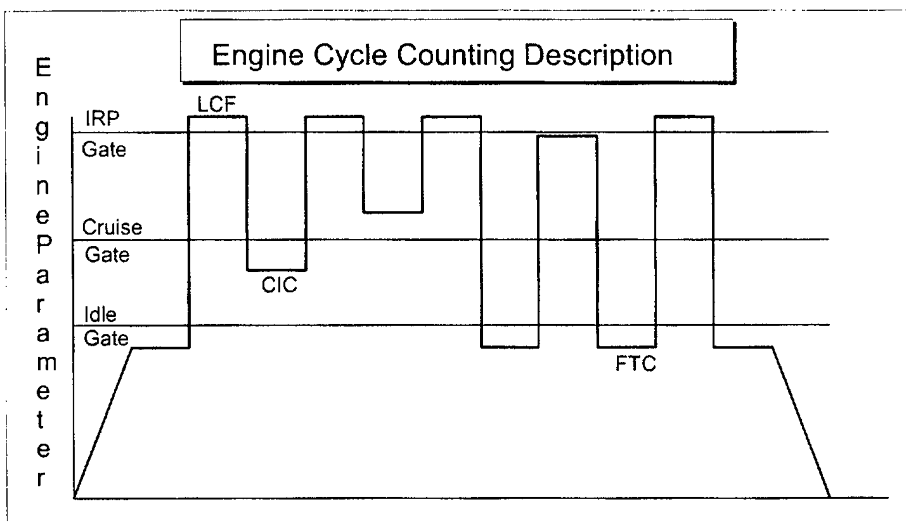



Figure 2 – TAC cycle definitions

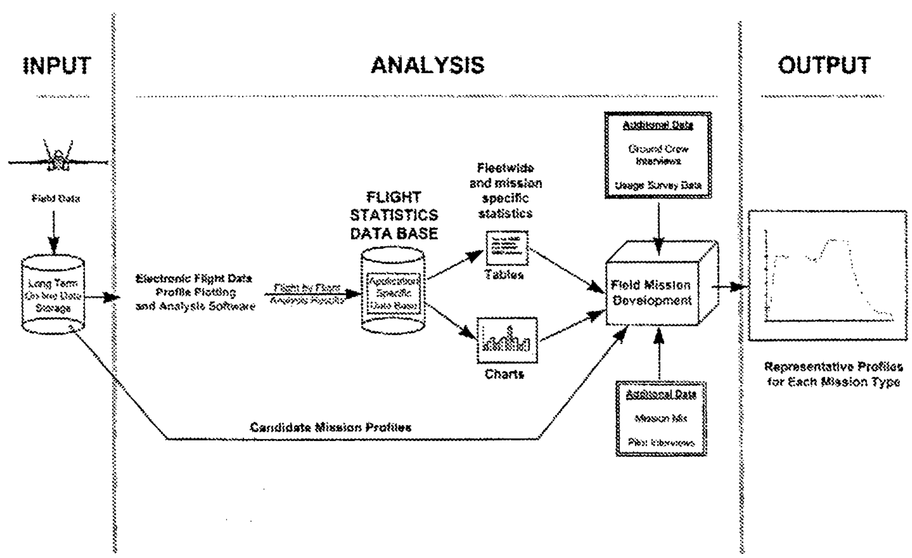



Figure 3 – TAC calculation

In order to prepare the data needed to conduct the life analyses, the missions are smoothed to minimise the number of points in each sortie profile, and the cyclic content and times are adjusted during this process. As a final check of the

resultant missions, their damage is compared to the damage of the actual missions to assure that it is representative of fleet experience. Finally, thermodynamic parameters are calculated for each point in the model mission profile, which covers all required deterioration levels and ambient conditions, and detailed transient profiles are provided for each modelled mission type.

The field limits for each component are then related to the engine field tracking statistics to assure consistency in the resulting field limits with the parts life tracking system data.

# Appendix 2
## Maintenance Policies and Procedures

by

*(O. Davenport)*

## 1. Overview of Current Maintenance Strategies

This section provides information that is current at the publication date, and does not reflect advances that will surely occur for the newest engines including the F119-PW-100, F119 derivatives, F120-GE-100 and the EJ200.

## 2. Canada - Canadian Forces

Canadian Forces (CF) engines are lifed based on the manufacturer's recommendations with a few exceptions. The CF policy on equipment lifing does not specify detailed lifing approaches, but rather provides broad guidelines to the equipment manager who is ultimately responsible for approving component lives. In general, the dynamic (rotating) component philosophy is a safe life approach. The engine types have been divided into three main categories: Transport/Patrol, Helicopter, and Jet.

| ENGINE TYPE | LIFING STRATEGY |
|---|---|
| **TRANSPORT** | |
| T56<br>(This is the main transport and patrol engine. The A310 and CF6-80 are maintained under civilian regulation) | All CF T56 engine components are life limited by inspection wear limits. However, the engine manufacturer has published LCF/Creep limits that are used by civilian operators. The USAF has indicated that it is in the process of implementing modified versions of these lives. Therefore, the CF are re-evaluating their approach and have already implemented a 23,000-hour life on the Series II 2/3 turbine spacer based on failures and destructive component testing. This value will probably be lowered, based on further testing and manufacturer/USAF recommendations. |
| **HELICOPTER** | |
| PT6T | All component lives are based on manufacturer safe life specifications (LCF limited). Currently lives are tracked on operating hours, but investigation into using cycle counts (mainly LCF) on the newer Bell 412 (CH146) aircraft with its HUMS recording capability, is taking place. |
| T58 | Components are lifed, based on manufacturer safe life recommendations. All CF T58 are in process of being upgraded to the -100 series and life limits will correspondingly change to the new model values. |
| **JET** | |
| J85-CAN40 | Component lives were established based on the manufacturer's safe life limits and CF Mission Severity Factors determined in consultation with the manufacturer. Recently, an engine speed monitor has been installed on the CF Tutor aircraft. This is to more accurately track cycle counts, and to establish more realistic conversion and severity factors. The CF also participates in a Component Improvement Program (CIP), where the manufacturer provides lifing updates to the entire user community. Some damage tolerance analysis has been completed for some components, and safe inspection intervals determined. |
| NENE X | The older NENE X has a combination of lifing approaches. The compressor impeller, turbine disk, and turbine shaft are removed from service based on inspection criteria. Additionally, the compressor impeller is safe life limited based on LCF cycles translated to operating hours. The last complete review of the conversion was done in the early '70s. Work is currently underway to install cycle counters to track actual usage of all components and update conversion rates. Consideration is being given to using cycles rather than operating hours for limits. A fracture failure analysis of the critical components is underway to better characterise component condition against usage. Some analysis of damage tolerance capability has been carried out for some components, and safe inspection intervals determined. |
| F404 | One of the newer CF engine fleets, the F404 is maintained under a modular concept with a comprehensive recording system. This monitors and tracks individual cycles based on manufacturer's algorithms and the manufacturer's safe life limits in cycles for LCF and thermal cycles. A computerised configuration management system oversees all 26 critical components plus another 135 tracked components. The CF also participates in the F404 CIP, which continually works with the manufacturer and other users to improve lifing methods and limits. |

## 3. FRANCE

| Engine Type | Lifing strategy |
|---|---|
| **JET** | |
| Larzac<br>(Alpha jet) | Component life is based on LCF cycles. Monitoring of engine life consumption is performed through Engine Flying Hours and the declared flight profile. The conversion factor between EFH and LCF cycles is calibrated for different flight profiles. For this purpose, some engines were equipped with tape recorders to monitor flight parameters. The damage for each flight is calculated using a ground-life-monitoring system. Life limited parts are scrapped once the life limit has been reached. |
| Atar 9K50<br>(Mirage F1) | Component life is based on LCF cycles. Monitoring of engine life consumption is done through Engine Flying Hours. The conversion factor between EFH and LCF cycles is calibrated for different flight profiles using the crash recorder. The inspection frequency is based on an EFH limit. Life limited parts are scrapped once the life limit has been reached. |
| M53-P2<br>(Mirage 2000) | Component life is based on LCF cycles. A life limit is set for each critical component. Each engine is monitored by an on-board life monitoring system that calculates the fatigue, the creep damage and the residual life for about 20 critical part locations. The authorised life is released after inspection of the fleet leaders to detect other possible damage mechanisms. The initial life, the residual life and the inspection frequency are expressed using the same flight damage unit limit. The statistics of residual life and spares forecasts are established on-ground using the data transferred from the on-board life monitoring system. |

## 4. GERMANY

| ENGINE TYPE | LIFING STRATEGY |
|---|---|
| **JET** | |
| J79 - 17A<br>(Phantom) | Part's lives are based on LCF cycles. Life consumption is monitored using engine flying hours (EFH). Cyclic exchange rates are provided by the engine manufacturer.<br>Fixed intervals (in terms of EFH) are defined for engine overhaul. During overhaul, parts are inspected for cracks, wear, corrosion, erosion, FOD and damaged coatings. Damaged parts are either repaired or retired. Parts with sufficient remaining life are re-used. |
| RB 199<br>(Tornado) | Part's lives are based on LCF cycles that are established on analysis and tests predicting minimum material properties. Lives are released in steps supported by a sampling programme. For critical areas with high damage-tolerance capability, lives are extended into the safe crack-propagation regime.<br>Technical Life Reviews are performed to check life assessment and assumptions for validity.<br>Life consumption is individually monitored with an On-board Life Usage Monitoring System (OLMOS), using sophisticated thermo-mechanical engine models covering fatigue and creep life consumption.<br>Life limited parts are inspected as they become available. Only parts with sufficient remaining life are re-used. |
| RD 33<br>(MIG 29) | Part's lives are based mainly on wear, corrosion and erosion. LCF is not life limiting. Lives are established on analysis and accelerated mission testing. Life is released in terms of engine running hours, where the appropriate scaling factor between test time and engine running time was originally established by the manufacturer.<br>A national life-extension programme (in co-operation with the engine manufacturer) was launched. With reduction of load and temperature, evaluation of German mission profiles and on-condition monitoring and inspections, the released life in terms of engine running hours could be significantly extended. |
| **HELICOPTER** | |
| 250 - C20B<br>(BO 105) | Part's lives are limited on either engine running hours or the number of engine starts, whichever is reached first.<br>Engine overhaul is performed at fixed intervals. During overhaul, parts are inspected for cracks, wear, corrosion, erosion, FOD and damaged coatings. Parts with sufficient remaining life are re-used. |
| T 64 - 7<br>(CH 53) | Part's lives are based on LCF cycles. Life consumption is monitored using engine flying hours (EFH). Cyclic exchange rates were provided by the engine manufacturer. Based on recorded German mission profiles, updated exchange rates have been defined.<br>Fixed intervals (in terms of EFH) are defined for engine overhaul. During overhaul, parts are |

| | |
|---|---|
| | inspected for cracks, wear, corrosion, erosion, FOD and damaged coatings. Damaged parts are either repaired or retired. Parts with sufficient remaining life are re-used.<br>When cracks were detected at unusual locations, they were reported to the manufacturer and led to new assessment of the parts and revision of the released lives. |
| T - 62T - 27<br>(APU in CH 53) | Component lives are limited by engine starts. The part's lives were originally specified by the manufacturer, but then drastically reduced during service leading to a high number of life-expired parts.<br>In a national programme, updated life limits were established based on actual operating conditions. |
| **TRANSPORT** | |
| Tyne Mk 22<br>(C 160 Transall) | Part's lives are limited by LCF cycles. The manufacturer increased the initial lives, as the result of a Life Development Programme.<br>Cycles are counted as 1 cycle per landing, 1/2 cycle per touch and go, no cycles per engine ground run. Average cyclic exchange rates have been established based on this counting practice and German mission profiles.<br>Engine overhaul is performed at fixed intervals (in terms of EFH). During overhaul the parts are inspected and - based on the findings - re-used or scrapped. Normally, engines are rebuilt with remaining lives sufficient for the next inspection interval. In rare cases, a reduction of the interval due to life limitations is accepted. |
| Tyne Mk 21<br>(Breguet Atlantique; Sea Surveillance, similar to transport) | As for Tyne Mk 22, but cyclic exchange rates and inspection-intervals are different. |

## 5. Greece - Hellenic Air Force

| ENGINE TYPE | LIFING STRATEGY |
|---|---|
| **TRANSPORT** | |
| T56 (C-130) | These are life limited by engine operating time. Life usage counters are currently being fitted in a number of aircraft (sample). |
| **HELICOPTER** | |
| T-53 | Life counters have been retrofitted into a number of engines and exchange rates, applicable for the whole of the engines, are statistically calculated. Life limits are based on the manufacturer's recommendations. |
| **JET** | |
| J69 | Life limited parts are retired on a flight time basis. |
| J85-4/13 | The lives are based on the manufacturer's, the USN's, and USAF's recommendations. They are in cycles. HAF mission severity factors for both the versions are statistically defined. Counters have been retrofitted in a number (sample) of engines. |
| J79 | Parts lives are based on the manufacturer and USAF's (given that USAF uses it) recommendations and a conservative exchange rate is applied. Studies are currently being undertaken to define the feasibility of retrofitting life usage counters. |
| TF-41 | Flight hours are used to track engine-component life-consumption. The limits are based on hours and are provided by the USN. |
| F110 | The lives are based on TAC's. The manufacturer's recommendation is used. Individual engine tracking is adopted and some of the engine rotating-parts are retired as soon as the life limit is reached. Others are inspected and cleared for a new period of use. |

## 6. Italy - Aeronautica Militare Italiana (AM)

In the past Turboprop, Helicopter and Jet Engine Manufacturer's recommendations were accepted, as engines were build or assembled under a licence agreement. The Italian Air Force (IAF) Component Lifing Philosophy has evolved in the last few years. This evolution began with the application of a sophisticated methodology for Data Recording, Monitoring and Tracking (Maintenance Recorder). It became possible to track individual component's life consumption in cycles and to analyse residual life. This can be performed via two methodologies, the first is the instrumentation of few aircraft that will be used as fleet leaders, the second is the installation of a recording device on the full fleet. Specific information

for engine type is provided in the following table.

| ENGINE TYPE | LIFING STRATEGY |
|---|---|
| **TURBOPROP** | |
| T64<br>(Aeritalia G222)<br>T56<br>(C130H)<br>TYNE<br>(Breguet Atlantique;<br>Sea Surveillance) | Originally, for all IAF Transport Aircraft Engines (T64, T56 & TYNE) belonging to the old 'generation', life limited components were subject to periodic inspections in term of operating hours and by inspection wear limits. This approach has been superseded by the introduction of a 'fixed' algorithm, given by the engine manufacturers, which allows conversion of operating hours to cycles. A further improvement has been the 'variable' algorithm philosophy approach. This derives a value directly from a table, originated by engine manufacturers. All mission profiles flown in a given period of operation are recorded and analysed to obtain the Average Life Distribution Consumption Corrective Factors (Different factor for each spool). The resulting factors, which will be subject to further regular updating, are used in the conversion algorithm. The IAF and Engine R&O company are also involved in the T64 Component Improvement Program, in which the engine manufacturer distributes lifing data updates to all operators.<br>The engine is maintained with an 'Overhaul at fixed intervals philosophy' (in terms of EFH). Parts are re-used in accordance with the above-mentioned process. |
| **HELICOPTER** | |
| | Group 'A' component lives are based on engine manufacturer's safe life limits and recommendations. Engine-component life consumption is recorded and parts are replaced accordingly to the engine manufacturer's instructions. |
| T700-T6E | Component lives for this engine, in the NH90 helicopter, will be set based on the engine manufacturer's safe life specifications (LCF). Lives and cycle consumption will be tracked via the EMA (Electronic Maintenance Assistant) that is part of the EECU (Engine Electronic Control Unit). Maintenance actions will be driven by the EMA. |
| **JET** | |
| J79 | For this engine, that belongs to the old 'generation', life limited components are subject to periodic inspections in terms of operating hours. In J79 CIP the engine manufacturer initially provided a 'fixed' algorithm that allowed the conversion of operating hours to cycles. Subsequently, 'variable' algorithms that provided Average Life Distribution Consumption Correction Factors were released. These will be subject to further updating on a regular basis, and are used in the conversion algorithm.<br>The engine is maintained with an 'Overhaul at fixed intervals philosophy' (in terms of EFH). Parts are re-used in accordance with this process. |
| SPEY MK807 | The SPEY fleet is currently maintained with an 'Overhaul at fixed intervals philosophy' (in terms of EFH). In this programme, to calculate component LCF in accordance with IAF Mission Severity Factors, two lead aircraft fitted with full Flight Test Instrumentation (FTI) and On-board Data recording Unit, have flown all Staff Requirement Missions in all foreseen profiles. The engine manufacturer provides a comprehensive and traceable algorithm that is applied to the full fleet. |
| RB199 Mk101 /103 | RB199 modular engine maintenance is supported by a dedicated system (MA.RE.-Maintenance Recorder) that records engine function parameters (throttle movement, duration, temperature, rotors speed. and thermal cycles) against all critical plus several major components (more then 100 items). The system, which equips the whole IAF fleet, has an On board Data Recording Unit and a Ground Station. All recorded flight data are downloaded into the Ground Station, which maintains the Engine configuration record and performs life consumption calculations for the individual components installed on that aircraft/engine and allocates subsequent cycle debits to the tracked components. The algorithms and safe life limits contained in the ground station are based on engine manufacturer data. The information retrieved from the system is also used in the life improvement programmes. |
| PEGASUS | Due to the uniqueness of this application (V/STOL A/C) and non-availability of the on-board recording unit, the Italian Navy Pegasus fleet is operated under the 'Overhaul at fixed intervals philosophy' (in terms of EFH). The component lives have been established in accordance with the engine manufacturer's |

| | |
|---|---|
| | recommendations.<br>Parts are inspected at overhaul, and replaced or repaired and re-used if sufficient life is still available. |
| EJ200 | The EJ200 engine, like the RB199 fleet, will be maintained in accordance with its modular design. However, as the previous systems revealed operational difficulties during extended deployment away from the Ground System, leading to a loss of cycles, it will be supported with an enhanced system being developed and applied to the whole EF2000 fleet. This system will elaborate and update the recorded data in near real-time, after engine cut-off, via the On-Board Processing Unit. This will avoid any data loss and subsequent analytical calculation in the ground station. |

## 7. NETHERLANDS

| ENGINE TYPE | LIFING STRATEGY |
|---|---|
| **JET** | |
| PWA F100-200/220 | Engine component lifing of this engine, installed in RNLAF F-16 fighter a/c, is based on the manufacturer's Damage Tolerance (Retirement-for-Cause) recommendations. Maintenance schedules and replacements are specified in terms of factored engine cycles. A separate Load and Usage Monitoring programme is performed. In this programme the high-pressure rotor speed N2 and the power lever angle PLA are being monitored, combined with a variety of relevant aircraft and flight parameters. The monitored data is transferred to the manufacturer that can use this data to update the maintenance schedule. |
| **HELICOPTER** | |
| Rolls-Royce Gem42 | Engine hard time maintenance of the Gem42 engine, installed in the Royal Netherlands Navy (RNLN) WHL Lynx naval helicopter is being applied on the basis of the so-called Cyclic Life Control (CLC) concept. Since 1991, the RNLN has applied CLC on a sample-monitoring basis, with 4 cycle counters measuring engine usage on a rotation basis within the fleet of 22 helicopters. Recently, RNLN has decided to switch to fleet-wide CLC. For this purpose a new fully autonomous data-acquisition system has been developed, built, installed and operated by the RNLN. This system combines engine usage monitoring with structural load and usage monitoring. To date, the RNLN is the only Gem operator to apply the Rolls-Royce designed CLC approach. |

## 8. NORWAY

| ENGINE TYPE | LIFING STRATEGY |
|---|---|
| **JET** | |
| J-85-GE-13 | Component lives were established based on OEM, USAF safe life limits and RNoAF Mission Severity Factor determined in consultation with the OEM/IEMP (International Engine Management Program). |
| F100 | The lives are based on TACs. The original engine manufacturer and USAF recommendations are used. Individual engine tracking is adopted and some of the engines rotating parts are retired as soon as the life limit is reached. Others are inspected and cleared for a new period of use. |
| **HELICOPTER** | |
| PT6T-3B | All components lives are set based on OEM safe life specifications (LCF limited). |
| GNOME H1400 | Component lives are established based on OEM recommendations. Lives are set based on LCF cycles converted to operating hours. The last complete review of the conversion factor was done by RNoAF in 1995 (1:1). |
| GEM 42 | The lives of all accessories are on condition. The lives of rotating parts/components are based on original engine manufacturer and are given in cycles or hours NRoAF use hours. |
| **TRANSPORT** | |

| **TURBOPROP AIRCRAFT** | |
|---|---|
| RR/Allison T56 Engine | Engine components are scrapped as a result of findings during scheduled inspections - no LCF life limits are yet established. However, we are aware of the manufacturer's LCF and creep limits published for commercial operators, and are currently evaluating the USAF and USN approach in this matter. It will be decided whether the USAF and USN policy should be applied, with or without adjustments to the RNoAF fleet.<br>RNoAF operates this engine in both the C-130 and P-3 aircraft, and has established a overhaul interval of 5500 flight hr and 6000 flight hr respectively for these engine installations. |
| P&WC PT6A-20 | All components are lifed based on the OEM recommendations. For rotating parts, both operating hours and cycles are monitored. |
| **TURBOFAN AIRCRAFT** | |
| AlliedSignal TFE-731 | RNoAF operates three Dassault Falcon DA-20 aircraft, which have recently been re-engined with the TFE 731-5BR.<br>All components are lifed based on the OEM recommendations. Engine performance and health are being monitored through a 'jetcare' program, where essential engine parameters are being recorded and analysed. |

## 9. PORTUGAL

The Portuguese Air Force (PoAF) follows the recommendations of the engine manufacturer to monitor the aging of their engines. For some, the PoAF has developed a 'health monitoring system' in co-operation with technical laboratories. The 'maintenance concept' is based on the manufacturer's recommendations and is supported by a set of adequate technical publications.

| ENGINE TYPE | LIFING STRATEGY |
|---|---|
| **JET** | |
| LARZAC<br>(Alpha Jet) | LARZAC is a modular engine. The module lives are based on OEM (Turbomeca) safe life limits and are function of the mission profiles. The control is done by cycles and by flight hours. For these particular engines PoAF has, in parallel, a health monitoring system in co-operation with a technical laboratory. The results are accurate. |
| TF30<br>(A-7P-Corsair) | TF30 components lives were established based on OEM safe life limits. The PoAF has developed a program to track the rotating components and performs the rework of the HP turbine at 900 hours and the high pressure compressor and turbine at 1500 hours. |
| F100<br>(F16) | F100 is a modular engine. The engine condition is monitored by the 'Engine Monitoring System' (EMS) which integrates an 'Advanced Compact Engine Tracking System' (ACETS). |
| TFE 731<br>(Falcon 50) | TFE 731 component lives are established based on OEM recommendations. The condition is controlled by an 'Engine Tracking System.' The maintenance concept is in accordance with civilian regulations. |
| **TURBO-PROP** | |
| T56<br>(C130-Hercules)<br>(P3-Orion) | All the T56 engines are life limited by inspection wear limits. For some, specific components are undertaken studies in order to determine if it is possible to rework them to continue in service. |
| TPE 331-251C<br>(Casa 212) | PoAF follows the OEM recommendations and cycles and flight hours control the parts. The hot section has a hot-end life inspection (HIS) that may lead to an extensive or reduction of the OVHL time. |
| **HELICOPTER** | |
| ARTOUSTE<br>(AL III-helicopter) | For this old engine, the components are removed from service based on inspection criteria. No rework is performed. |
| MAKILA<br>(SA 330-PUMA Helicopter) | MAKILA is a modular engine. The modules lives are established based on OEM (Turbomeca) safe life limits. |

## 10. Spain

| ENGINE TYPE | LIFING STRATEGY |
|---|---|
| **JET** | |
| J-79 | Maintenance according to USAF T.O. 2J79-83 and T.O. 2J79-86 manuals and updates proposed by the Technical Integrated Group for Engineering and Reporting Support (TIGERS) from General Electric.<br>There is a Component Replacement Program recommended by TIGERS group, which consists of the systematic replacement through overhaul of a set of 23 elements. From all these elements, the replacement of the Outer Combustion Cases (OCCs) is actually being performed, as it has been considered a very critical element, due to the loss of some aircraft in other countries. In regard with the rest of the elements, the critical elements will be replaced for Flight Safety. |
| F404 | The GE F404 is a modular engine, with fully interchangeable components, with an 'on condition' maintenance philosophy. This is controlled solely by the accrued fatigue induced by thermal stress, pressure, revolutions and operating hours on those parts that are limited by Low Cycle Fatigue. The Lifing team in GE has established fatigue limits for the parts. Each component is assigned a fatigue limit (LCF), according to the its' failure probability and criticality.<br>Based on the studies of the manufacturer (GE), through finite element analysis, endurance test, etc., the failure manner, appearance and propagation rate of the crack and consequences of failure have been determined. With this information all critical elements have their lives tracked (accrued fatigue). An automatic system records the parameters causing the stress for each flight. These are processed through an ADP (Automatic Data Processing) and accrued life counts are assigned to each component. The algorithm is provided by GE. It uses the average mission mix of each fleet.<br>SAF, currently, is in the process of applying a specific algorithm for each type of mission and part, trying to optimise the life calculations. SAF has accomplished simulations with this new algorithm that demonstrate in the case of the Spanish fleet an average saving of 7% in the life consumed with respect to the system currently used by other fleets. |
| F404 continued | The life limits are checked periodically and revised in the light of experience, removed parts analysis, new engineering studies, pressure failure studies, specific trials, and through programs of Life Validation. Within the life validation program are found the 'Fleet Leader', the engine analytical condition inspection (ACI), the improvement components program (CIP), etc. Due to the disparity in the life limits of each part, numerous engine disassemblies are produced during the engine life.<br>SAF based on economic and operational concepts, has established the concept of 'Window' grouping within a margin of LCF lives, to avoid unnecessary dis-assemblies and to maximise engine availability at the lowest possible cost. This system compels us to have sophisticated integrated logistical support, to adapt the system to the continuous life limit modifications. The support system is a function of the operative needs of the fleet, availability of human resources, parts, facilities, economic, statistic, capacity of the difference maintenance levels, industry response time.<br>SAF accomplishes simulations that help achieve the designed operational objectives, in the most efficient way. These simulations are accomplished by considering the future evolution of the fleet. This allows planning of the necessary resources, and the prediction and avoidance of problems. |
| **Turbo-Prop** | |
| T-56 | 1. Maintenance program for T.56-A-14<br>Maintenance is 'ON CONDIITON' according to NAVAIR 01-75PAA-, 02B-5DD manuals and OPNAVINST 4790.2 general instructions.<br>PERFORMANCES NA01-75PAA-6 is basically as follows: overhaul of injectors every 4,800 hours; replacement of the turbine rotor at 7,500 hours |

| | |
|---|---|
| | (35,000 hours for some P/N); withdrawal of the turbine stator at 35,000 hours (7,500 hours for some P/N).<br>Engine inspection periods are:<br>28 days - special inspection of aircraft and engine.<br>112 days - special inspection of aircraft: Basically engine corrosion<br>224 days - special inspection of aircraft: Engine corrosion, lubrication, controls and cleaning<br>150 hours - special inspection: Engine efficiency;<br>300 hours phased inspection of aircraft: Deep inspection of engines; conditional inspections, etc.<br>Maintenance Program for T.56.-A-15<br>Maintenance according to USAF T.O. 2J-T-56-53 and T.O. 2J-T-56-56 manuals and O.T.E.-2J-T-.56-2 and O.T.E.2J-T-56-3 manufacturer (Allison) manuals. |
| J85-GE13 | Maintenance for J85-GE13 engines, which constitute the Air Force inventory, complies with XX2F-5A-6WC-4 work cards and procedures issued by the USAF.<br>These work cards specify a maintenance period of 300 hours, so every time the engine accumulates this number of operating hours, it must go through the workshops. Several maintenance activities will take place, including overhaul, repairing or replacing different parts of the whole. The depth of inspection or revision work increases such that the engine dismount level progressively increases, until it covers a period of 2400 hours, when the engine is completely dismantled. After this, a new maintenance cycle begins.<br>Most engine elements are 'on condition'. Depending on their condition when dismantled during programmed inspections they will be replaced or repaired, in conformity with limits specified by the applicable Technical Order issued by the USAF. Certain 'fungible' elements as well as engine ignition plugs are replaced when inspections are carried out.<br>Some of the components have an LCF limited operating life. Replacements have been phased into the scheduled maintenance plan, where possible. Only components for the impellers and turbine rotor (discs and spacers) are excluded.<br>The severity of aircraft missions is recorded via operating hours counters. Engine operating hours can be converted in fatigue cycles. Replacement of parts that exceed the operating life periods is done at the programmed inspections closest to the expiration time. |

## 11. TURKEY

Turkish Air Force engines are lifed in accordance with the original engine designer and manufacturers' recommendations. The dynamic component philosophy is mainly a safe life approach. There are some exceptions.

The Turkish Air Force manages aircraft engines through a weapons system approach. In this management style, the policy does not specify detailed lifing approaches, but rather provides broad guidelines to the weapon system (engine) manager who is ultimately responsible for approving component lives.

Specific information for each engine type is provided in the following table. The engine types have been divided into four main categories: Turbofan, Turbojet, Turboprop, Turboshaft, and Auxiliary Power Units.

| ENGINE TYPE | LIFING STRATEGY |
|---|---|
| **TURBOFAN ENGINES** | |
| F110-GE-100/129 | Employed maintenance policy is ENSIP. Engine parts related to the ENSIP interval are inspected and cleared for a new period of use. Some of the rotating parts are retired as soon as the life limit is reached.<br>Parts' lives are based on TAC, which is a combined measure of partial and full LCF cycles accumulated by engine. For parts lifing, original equipment manufacturer's (OEM) recommendations are used. |
| CFM56 | Current lifing policy is strictly in line with OEM's publications. |
| **TURBOJET ENGINES** | |
| J69 | A study is being carried out by TuAF to define a new maintenance concept and critical life limits according to inspection results of previous years. |

| | |
|---|---|
| | Fixed intervals (1000 EFH) are defined for engine overhaul. During overhaul, parts are inspected for FOD, wear, damaged coatings, erosion and corrosion. Damaged parts are either replaced or repaired.<br>Used life of the life-limited parts is tracked by the engine flight hour. |
| J79 | Fixed intervals (1200 EFH) are defined for engine overhaul. USAF Technical Order uses number of sorties for overhaul period. A correlation study was accomplished to define the engine flying hours limit. During overhaul, parts are inspected for FOD, wear, damaged coatings, erosion and corrosion. Damaged parts are either replaced or repaired.<br>Both USAF and OEM's recommendations are taken into consideration for the life limit of the engine life limited parts. But; limits are subject to change based on NDI results of overhaul.<br>TuAF also participates in the J79 TIGERS (Technical Integration Group for Engineering and Repair Studies) which continually works with the manufacturer (General Electric Aircraft Engine) and other users to provide lifing updates, improve limits and share best practices. |
| J85 | Fixed intervals (in terms of EFH) are defined for engine overhaul. TuAF defined the interval and scope of the overhaul, in accordance with experience. This maintenance concept is called 'Extended Depot Level Maintenance' in which base level responsibilities are minimised and modular maintainability is also included. During overhaul parts are inspected for FOD, wear damaged coatings, erosion and corrosion. Damaged parts are either replaced or repaired.<br>Part lives are based on USAF recommendation. Some counters (Engine Life Monitoring, ELM) have been retrofitted in a number of engines to be able to calculate an d compare the mission severity factors of different wings and bases. Life limited parts are tracked by EFH and limits are totally defined by past NDI history. OEM support is also used for the analysis of NDI results and risk assessment. |
| **TURBOPROP** | |
| T56 | Engine overhaul is performed at fixed intervals (in terms of EFH). During overhaul the parts are inspected for FOD, wear, damaged coatings, erosion and corrosion. Based on findings, damaged parts are repaired for re-using or replaced. Part lives are based on inspection wear limits, which are recommended by OEM.<br>TuAF also participates in the T56 CIP which continually works with the manufacturer (Allison Engine Company) and other users to provide lifing updates, improve limits and share best practices. |
| Tyne Mk 22 | Engine overhaul is performed at fixed intervals (in terms of EFH). Although the OEM's recommended life limits are based on cycle records, TuAF has defined the conversion rate of EFH through a statistical study. During overhaul the parts are inspected for FOD, wear, damaged coatings, erosion and corrosion. Based on findings, damaged parts are repaired for re-using or replaced.<br>Parts have cyclic consumption are defined by the OEM and called Group 'A' components. Group 'A' component' lives are based on OEM's safe life limits and recommendations. Engine component life-consumption is recorded and parts are replaced according to the OEM's instruction.<br>TuAF also participates in the technical support program with MTU-Germany and other users to provide lifing updates, improve limits. |
| CT& | An on-condition modular-engine-maintenance concept is used.<br>Part's lives for this engine are set based on engine manufacturer safe-life specifications. Cycle consumption is tracking via electronic engine control unit. |
| **TURBOSHAFT ENGINES** | |
| T700 | An on-condition modular-engine-maintenance concept is used.<br>Part's lives for this engine are set based on engine manufacturer safe-life specifications. Cycle consumption is tracked via electronic engine control unit. |

## 12. UK - ROYAL AIR FORCE

The authorised life for a RAF aero-engine type or mark, its modules and accessories is decided and controlled by the appropriate Engineering Authority (EA) with advice from the manufacturers and the MOD(PE), and published through the MOD(PE) Local Technical Committee (LTC) at the manufacturers works. Other than for ancillaries, these lives are published in AP100E-01B. Lives for ancillaries are published in the Master Maintenance List (MML) for the aircraft. A variety of strategies has been adopted to produce authorised lives, the most common being a safe life determined from fatigue-based criteria, although the damage tolerant approach has been applied in a few cases.

| ENGINE TYPE | LIFING STRATEGY |
|---|---|
| **JET** | |
| RB199 (All Mks)<br>(Tornado - all Mks) | The RB199 is a modular engine with 22 Group A components, which are individually lifed in cycles based on the original manufacturers algorithms, and safe life limits in accordance with Tri-National Document TU346. These limits can either increase or decrease depending on a series of on-going spin tests to prove the LCF characteristics of each component. The rate of cyclic usage is calculated for each engine Mk from a small sample of calibrated engines flown through various sortie profiles.<br>The life of each Group A component is tracked with a computerised management system that is capable of recalculating total life based on previous usage. Additionally, specific component and fleet risk management is used as a short term management tool for damage-tolerant fracture critical Group A components. This allows component life to progress into the crack propagation phase to permit identified items to exceed their release to service life to minimise the effects of component life reductions. |
| Adour Mk15101<br>(Hawk - all Mks)<br>(The RAF's Advanced Flying/Tactical Training a/c) | All Group A components are hard lifed in cycles in accordance with the Life Management Plan. However, whereas several of the Group A components are subjected to some form of periodic inspection during their time in service, others serve out their 'one shot' life without being inspected. |
| Adour Mk15102<br>(Hawk for the RAF's Aerobatic Team, The Red Arrows) | As per the Mk15101, however, the Group A component lives and exchange rates have been factored to compensate for the demanding aerobatic role of the aircraft. |
| Adour Mk10401<br>(Jaguar GR1, GR1A, T2 - low-level, ground attack and reconnaissance a/c) | As per the Mk15101, however, owing to the role of the aircraft and the fact that the engine uses reheat (after-burner), the Group A component lives are subject to factored exchange rates. Further, given the nature of some GR1/1A out-of-area operations, the exchange rates are again factored and additional 'hot section' inspections are carried out. |
| Pegasus<br>(Harrier) | The Pegasus has 36 Group A components which have individual lives based on the original manufacturer's algorithms and safe life limits. Due to the specialist nature of the engine, the authorised component lives are expressed in a combination of fatigue counts, creep counts and thermal fatigue counts. |
| **SURVEILLANCE** | |
| CFM56-2<br>(Sentry AEW Mk1 a/c) | Engine life is calculated in flight cycles linked to Group A cyclic lives; although current lifing policy is strictly in line with CFM publications, the Company feels that the RAF are over recording and, therefore, wasting engine life. Discussions are on-going with the Company to recover previously over-recorded engine cyclic life. It is hoped that the results of a manual recording exercise will allow the Company to provide the RAF with a reduced cyclic recording penalty. |
| Spey Mk250/251<br>(Nimrod) | The engine/module has an overhaul life stated by the engine manufacturer. Furthermore, all Group A component lives are based on the manufacturer's safe life specifications (LCF limited). Currently, life is tracked in operating hours and factored via a cyclic exchange rate into usage cycles. There is currently an on-going program to prove/revise the existing cyclic exchange rate. |
| Avon<br>(Canberra) | The Avon is non-modular. All engine and accessory lives are set by Rolls-Royce at a Local Technical Committee (LTC), accepted by the RAF and published in the LTC Register and AP100E-01, although RAF policy is |

| | |
|---|---|
| | ultimately decided by the EA. Engines are lifed by flying hours, with a calendar back-stop. Internal engine components are separately lifed by flying hours. Remaining components are 'on-condition'. An exercise to monitor the cyclic exchange rates is in progress to confirm that they are correct. |
| **TRANSPORT** | |
| ALF502R-5<br><br>(BAe 146) | ALF502 Group A life limits, based on cyclic consumption, are currently being reviewed by DERA and Allied-Signal. The manufacturer's published cyclic lives are under discussion. The engine is operated under 'Power-by-the-hour' arrangements to an on-condition policy. |
| Conway Mk301<br><br>(VC10 all Mks) | The engine has an overhaul life stated by manufacturer. Furthermore, all Group A component lives are based on the manufacturer's safe life specifications (LCF limited). Currently, life is tracked in operating hours and factored via a cyclic exchange rate into usage cycles. There is currently an ongoing program to prove/revise the existing cyclic exchange rate. |
| RB211-524B4<br><br>(Tristar) | The RB211 engine is ultimately controlled by Group A life. Component lives are calculated by the manufacturer and published in the lifing manual. In-service results are used by the manufacturer to reassess the cyclic lives. |
| T-56-A-15<br><br>(C-130 Hercules) | Engine and accessory life policy is recommended by Huntings Airmotive, after consultation with Allison Engine Company, at an LTC meeting and published in the LTC Register. The EA then decides RAF policy, which is published in AP100E-01. All engines and accessories are lifed in flying hours, with a calendar back-stop for certain items. The remaining components are maintained 'on-condition'. |
| TFE731-3R<br><br>(BAe 125) | Engine life is ultimately controlled by Group A life. Engine component cyclic life is published by the manufacturer and monitored by the operator FRA Serco. The engine is operated under 'Power-by-the-hour' arrangements. |
| **TRAINER** | |
| TPE331-12B<br><br>(Tucano) | Overhaul life is as recommended by the manufacturer and based on operating hours. There is also an inspection of the 'hot section' at half life. |
| Viper Mk301<br><br>(Dominie) | Overhaul life is as recommended by the manufacturer, following life assessment, and is based on operating hours. |
| Allison 250-B17C<br>(Islander)<br>(*Augusta 109*) | Overhaul life is as recommended by the manufacturer and based on operating hours and starts. There is also a part life rework required at half life. |
| **HELICOPTER** | |
| T55-L-712F/714A<br><br>(Chinook) | Engine component lifing is based on the manufacturer's safe-life recommendations. At present the T55-L-712F component lives are under investigation by both the manufacturer and DERA Pyestock. DERA component life recommendations, which are considered very conservative, have been promulgated. Information gathered from the engine management system (FADEC) has been used to support the DERA findings and an internal component engine life assessment, the results of which are expected in January 1998. |
| Turmo III C4<br><br>(Puma) | Authorised engine life is as recommended by the manufacturer and based on operating hours. |
| Gem 10001<br><br>(Lynx) | Overhaul life is as recommended by the manufacturer and based on operating hours. |
| Gnome 122/3/4/5<br><br>(Sea King) | Overhaul life is as recommended by the manufacturer, based on operating hours and subject to a part life rework at half life. |
| Artouste Mk2C<br>(Alouette) | Overhaul life is as recommended by the manufacturer, based on operating hours. |
| Astazou (all Mks)<br>(Gazelle)<br>(*Jetstream*) | Overhaul life is as recommended by the manufacturer, based on operating hours |

# 13. US ENGINES

## 13.1. U.S. AIR FORCE

The USAF has been employing ENSIP maintenance policies since 1979 on the F100-PW-100/200 engines. This is a classical damage tolerance approach to life management. Critical rotating components, static structure, and pressure vessels are retired at the minimum predicted LCF life. Present exceptions are some components of the F100-PW-100/200 engines, which make full use of the damage tolerance philosophy by employing 'retirement for cause'. All critical hardware are inspected with eddy current, ultrasonic, or fluorescent penetrant inspection when they are procured from the OEM and then re-inspected at pre-determined intervals set by damage tolerance analysis.

All older engines, which entered service prior to the establishment of ENSIP policies, are managed similarly to the Navy. When problems such as LCF cracking, occur with these engines damage tolerance assessments are often used to establish a risk based life management approach. This is to ensure supportability and fleet readiness while maintaining an acceptable level of risk.

The USAF also procures transport and trainer aircraft that are FAA certified. For older engines, these are maintained just as their civil counterparts. However, newly developed engines require damage tolerance analyses and a damage tolerant life management approach for setting inspections prior to the FAA certified 'hard time' life limit.

## 13.2. U.S. NAVY

The US Navy maintenance strategy for critical rotating and pressure vessel components is to retire these parts at the minimum predicted safe-life (LCF) limit. Parts are inspected for manufacturing defects, with fluorescent penetrant (FPI), when they are purchased. Life limited parts are inspected during operational service when they are fully exposed at the depot for other maintenance purposes. This is known as an opportunistic inspection and is conducted using FPI methods. One part may be inspected many times within its service life while an identical part is never inspected. Whenever an LCF crack is found, the part is immediately retired from service and forwarded to the manufacturer for further investigation. A complete review of the lifing methodology, analysis and empirical data is performed to verify or validate the existing life limit.

All engine LCF life limited parts are managed to a hard time limit. This is true for turbo-prop, turbo-shaft, turbo-jet and turbo-fan engines. The majority of mature navy engine LCF limited parts are tracked by engine flight hours. Only the most recently designed engines are fitted with high fidelity engine monitoring systems to track mechanical speed cycles and other life limiting events.

Since 1993, the Navy life management organisation has initiated programs with most engine manufacturers and engine models to update existing life limits. The updates involve:

- Lifing to the flight profiles recorded during missions;
- Enhancing thermal and stress models using state-of-the-art analysis tools;
- Incorporating fully characterised materials curves;
- Substantiating model inputs using instrumented engine test results.

| ENGINE TYPE | LIFING STRATEGY |
|---|---|
| **Fighter** | |
| F402 | The F402 turbofan has 36 flight critical (group A) parts which are individually lifed in cycles based on the original manufacturers design and a composite mission representing Harrier operations. There is an onboard cycle counter that records eight parameters, yet the components are retired on flight hour limits. Twenty-four aircraft currently flying the -408 model are fitted with instrumented flight recorders to define the current mission profile. When the new mission is defined, component exchange rates based on thermal transient analysis will produce new life limits. A revised field management plan will be developed to adjust spares needed to maintain the Harrier readiness rates. Additionally, fleet risk management is used for damage tolerant parts allowing use of fracture mechanics life. The use of propagation life minimises the impact of parts life reductions when usage values increase and spares are inadequate. The F402 is managed with a Hot Section Inspection interval. |
| F404 | The F404 engine has 25 LCF flight critical parts that are tracked and managed to a cyclic limit. The In-flight Engine Condition Monitoring system (IECMS) records the complete mission as well as temperature, speed, and pressure counts. The Equivalent Low Cycle Fatigue (ELCF) of each rotating part is tracked as a function of engine parameters. A comprehensive Life Management Master Plan has been established to update life management activities for both F404-400 and -402 engines over the last five years. Mission updates have been completed in 1992 and 1997 with a complete reanalysis of part lives |

| | |
|---|---|
| | between 1993 and 1997. The F404 is a modular engine that is maintained under the RCM philosophy. |
| F405 | The F405 turbofan has 10 flight critical (Group A) parts, which are individually lifed and managed in cycles based on the original manufacturer's design and a composite mission representing the T45 operations. These limits can be increased or decreased when new spin pit or material test data becomes available. An update of the mission profile is currently underway using improved Automated Data Recording (ADR) systems installed on 7 fleet aircraft. When the new mission is defined, component exchange rates based on improved thermal transient analyses will be upgraded. A comprehensive Life Management Master Plan (LMMP) is being prepared that will detail the lifing methodology specific for this engine. All aircraft are outfitted with an ADRS which records eight parameters used for tracking life consumption. The engine is maintained under the RCM philosophy. |
| F414 | The F414 turbofan has 20 critical Low Cycle Fatigue (LCF) and damage tolerant parts. The design missions consist of 14 flight profiles, 2 ambient temperature conditions and 2 levels of engine performance deterioration. Life analyses were performed for each of these combinations of conditions and combined by weighted-average to obtain overall life predictions. Current strategy upon completion of E&MD is to fly for the cyclic equivalent of 4000 hours and introduce eddy current inspection at that point. Eddy current inspections will be performed at fixed intervals until the LCF life limit is reached. A full mission analysis update is scheduled for FY04. The F414 is a modular engine that is maintained under the RCM philosophy. |
| F110 | The F110 augmented turbofan has 22 flight critical parts, that are individually lifed and tracked using the USAF TAC methodology. Mission profiles were updated in the early 1990's via instrumented flight recorders. All components life limits were reanalysed between 1993 and 1997 using enhanced finite element modelling and heat transfer analyses. Another mission analysis is in progress to assess life impacts from increased mission roles and requirements. The F110 is a modular engine that is maintained under the RCM philosophy. |
| TF34 | The TF34 turbofan has 32 flight critical parts that are managed to a 3 sigma standard deviation life limit in hours. All critical parts are undergoing reanalysis with mission updates and enhanced analytical modelling. The life limit updates were initiated in 1994. The missions were updated in 1995 using in-flight data recorders and will be reviewed again in the year 2000. The engine is managed under the RCM philosophy with a hot section inspection interval. |
| J85 | The J85 turbojet engine has two versions flying in the US Navy fleet. A nine stage compressor in the -21 model and an eight stage compressor in the -4 model. The first is used in F-5, and the later is used in the T-2 primary jet-trainer. All critical parts are lifed in cycles and managed to flight hour limits. A life management plan has not been implemented for at least 5 years, and mission analysis has not been updated within 10 years. Current life reductions are proceeding and fleet logistics problems are anticipated. Evaluations of longer aircraft use are currently being considered. A Life Management Master Plan will be forthcoming. |
| **Helicopter** | |
| T58 | The T58 turbo-shaft engine has two versions in the US Navy fleet. The -402 model is used in the H-3 and H-46 and the -16 model is used in the H-46E. The Navy is currently updating part lives. The analysis should be complete by the end of CY98. The -16 task was started in FY98 and should be competed by FY00. Part lives are based on flight hours. The engine is maintained under the hot section inspection and overhaul philosophy. Mission updates are being planned using in-flight data recorders. A field management plan for parts that have a severe reduction in LCF life will be developed. Propagation life will be used to offset these reductions with careful attention to fleet risk and flight safety. |
| T700 | T700 turbo-shaft engine has 17 flight critical parts lifed in cycles based on the original manufacturer's design and H-60 specification mission profiles. All engines were outfitted with engine history recorders at production that track four parameters (LCF1, LCF2 and Time at Temperature Index and Engine Operating Time) even though life limits are maintained in flight hours. Analytical life updates were begun in 1993 and will be completed in the next five three years. Mission profiles are being updated using in-flight data recorders on fleet aircraft. The T700 is a modular engine maintained under the RCM philosophy. |
| T64 | The T64 turbo-shaft has five type/model/series in the fleet. The -413, -415,-416, -416A, and the -419 are all used in various versions of the H-53. The critical part life updates for the -416 has been completed. All critical parts except one are the same between all versions. Part |

| | |
|---|---|
| | lives are based on flight hours. The engine is maintained under the hot section inspection and overhaul philosophy. In future years, the Navy will update material testing, mission recordings, and instrumented engine tests. |
| T406 | The T406 turbo-shaft engine program is managed under the 'power-by-the-hour' concept based on commercial-engine maintenance-support experience. There are 21 flight critical rotating parts which are individually lifed in LCF cycles and equated to mission hours based on the original specification missions representing the V-22 Osprey operations. In addition to the 21 flight critical parts, also 8 blades and vanes are life limited by either stress rupture or hot corrosion. There are 15 other components tracked by operating time. The V-22 aircraft are equipped with on-board recording and monitoring system to gather engine usage and mission profiles. Once production engines are fielded, engine usage and mission profiles will be gathered and verified. These usage and profiles will be analysed and results will contribute to establishing updated life limits. |
| **Transport** | |
| T56 | The T56 turboprop engine has 7 critical rotating life-limited parts. Two different series of this engine are used in four different Navy aircraft platforms: P-3, C-130, E-2C/C+, and C-2A. LCF and FM lives have been established for these parts and are managed using flight hours based upon a cycle provided by the OEM. The mission analysis for all four platforms has not been updated within 10 years. Recorded E-2C+ data will be utilised to update the mission cycle to flight hour conversion factor. No life management plan has been developed or followed within the last 10 years. A new life management plan is being developed. Turbine spacer life limits are under review in light of new OEM materials data and life limit reduction recommendations. |

# Appendix 3
## Mechanics of Materials Failure

by
*(W. Beres)*

## 1. FAILURE MODES OF GAS TURBINE ENGINE COMPONENTS

The safe life approach has been criticised as being overly conservative and costly on the ground that critical engine components are usually discarded with significant amount of useful residual life. There are two major concerns when a cycle-to-crack-initiation rejection criterion is used to life rotating parts. The first one is that, by implication, 99.9% of the components will be retired before any detectable crack has formed. Possibly, these parts may be capable of significantly longer service before they develop a detectable crack (actually 0.8-mm in length). Secondly, the components may be capable of tolerating crack sizes much greater than the 0.8-mm mentioned above. This crack limit actually reflects the sensitivity and reliability of current NDI methods rather than the mechanical tolerance of a particular part for cracks. Crack detectability by dye penetrant methods depends on crack width, whereas crack length is the critical factor for lifing purposes.

In their most elementary forms alternative damage tolerance based lifing procedures, which are also known as Life-On-Condition, Retirement-for-Cause or simply Fracture Mechanics (FM) lifing, make assumptions. They assume that the fracture critical locations of a component contain cracks of a size that lie just below the detection limit of the NDI technique used to inspect the component. The crack is then assumed to grow during service in a manner that can be predicted by linear elastic fracture mechanics, or any other acceptable methods. This is applied until a predetermined dysfunction limit is reached beyond which the risk of failure due to rapid crack growth becomes excessive (Koul at al., 1988).

For damage tolerance based procedures to be successful, supporting methodologies must be developed. These methodologies include both deterministic and probabilistic fracture mechanics based life prediction, based on non-destructive inspection, mechanical testing of test coupons and components, structural analysis, mission profile analysis and condition monitoring of components. In particular, extensive materials testing involving room and elevated temperature crack growth rate data must be performed for the application of new concepts. Therefore basic properties of materials and databases of material degradations modes are summarised in this chapter.

Not all components are candidates for life usage management to damage tolerance limits. Two examples are:

- Components made out of material with a short crack propagation life;
- Components that cannot be inspected.

Few, if any, mature engines were designed with damage tolerance (FM) methodologies and some may be viable candidates for damage tolerance management. An example of where this approach has been introduced after the engine has entered service, is the RB199 engine in the Tornado aircraft. A number of components, which have proven crack resistance, have been reassessed using fracture mechanics methods.

## 2. STRESS STATES IN GAS TURBINE ENGINE COMPONENTS

### 2.1. THERMAL ANALYSIS

The temperature distribution for turbine discs depends on the engine operating conditions, and which phase of a flight or ground run the engine is in. For a stabilised high temperature condition, the rim of the disc is hotter than the bore area, which creates compressive hoop stresses in the rim and tensile stresses in the disc bore area. This state of stress is reversed during engine deceleration when the rim cools more quickly and a reverse temperature gradient is established. The rim is then subjected to high tensile hoop stresses, and the disc bore area to hoop compressive stresses. Differences between the temperatures on each side of the disc create stress gradients across the disc thickness, which may also be taken into account.

### 2.2. STRESS ANALYSIS

Generally, for the linear case, the state of stress at a given point in the disc can be described as follows:

$$\sigma_{ij} = B_1\,\omega^2 + B_2\,\Delta T + B_3\,T \qquad \text{(A-1)},$$

where $\sigma_{ij}$ is the component of stress tensor, $\omega$ is the rotational speed of the components, $\Delta T$ is the temperature range, $T$ is the local temperature and $B_1$, $B_2$, $B_3$ are parameters depending on the material properties and component geometry.

#### 2.2.1. STRESS INTENSITY FACTOR

Under linear-elastic conditions, the general expressions for the crack tip stresses have the form:

$$\sigma_{ij} = \frac{K}{\sqrt{2\pi r}}\, f_{ij}(\theta) + A_2\, g_{ij}(\theta) + A_3\, h_{ij}(\theta) \qquad \text{(A-2)},$$

where the notation presented in 1 are used. $K$ is the proportionality constant called the *stress intensity factor*, $r$ and $\theta$ are the radius and polar angle measured for the crack tip, and crack plane respectively, $A_i$ are constants, $f_{ij}(\theta)$, $g_{ij}(\theta)$, $h_{ij}(\theta)$ are dimensionless functions of $\theta$ and $(i, j=1,2,3)$. If higher order terms are neglected ($A_2=0$, $A_3=0$), the stress components ahead of the crack tip in Mode I can be approximated by the expressions:

$$\sigma_{11} = \sigma_x = \frac{K}{\sqrt{2\pi r}} \cos\frac{\theta}{2}\left[1 - \sin\frac{\theta}{2}\sin\frac{3\theta}{2}\right] \qquad \text{(A-3)}$$

$$\sigma_{22} = \sigma_y = \frac{K}{\sqrt{2\pi r}} \cos\frac{\theta}{2}\left[1 + \sin\frac{\theta}{2}\sin\frac{3\theta}{2}\right] \qquad \text{(A-4)}$$

$$\sigma_{12} = \tau_{xy} = \frac{K}{\sqrt{2\pi r}} \left[\sin\frac{\theta}{2}\cos\frac{\theta}{2}\cos\frac{3\theta}{2}\right] \qquad \text{(A-5)}$$

These equations describe the stress singularity around the crack because as r→0, the stresses approach infinity. Thus,

stress near the crack tip varies with $1/\sqrt{r}$, regardless of the

configuration of the cracked body. It can also be shown that crack opening displacement near the crack tip varies with $\sqrt{r}$ .The stress intensity factor (SIF), $K$, is usually given a subscript denoting the mode of loading, i.e. $K_I$, $K_{II}$, $K_{III}$. It was found that the SIF range is a main factor governing the rate of growth of fatigue cracks. The SIF defines the amplitude of the crack tip singularity. This means that stresses near the crack tip increase in proportion to $K$. In addition, the SIF defines completely the stress conditions at the crack tip. If $K$ is known, it is possible to solve for all components of stress, strain and displacement as a function of $r$ and $\theta$. This is one of the most important concepts in linear elastic fracture mechanics: having one single parameter that completely describes the conditions around a crack tip. A distribution of an elastic stress $\sigma_y$ in front of a crack is shown in figure 2.

To make SIF a useful parameter, it should be possible to determine $K$ from remote loads and the geometry of a cracked component. Closed form solutions for SIF have been derived for a number of simple configurations. For example, for an infinite plate under tension treated in 2D, figure 3, the stress intensity factor at the tip of the crack is:

$$K_I = \sigma_y \sqrt{\pi a} \qquad \text{(A-6)},$$

where $\sigma_y$ is the remote stress and $a$ is the half of the crack length.

This equation is modified when an edge crack in a half plane is considered, figure 4:

$$K_I = F_I \sigma_y \sqrt{\pi a} \qquad \text{(A-7)},$$

where $F_I$ is the boundary correction factor, which takes into account the effect of free boundaries on the crack opening. For example, for a short edge crack in a plate $F_I$ = 1.12.

When considering three dimensional cracks, the closed form solution for a circular (penny shaped) crack fully embedded in an infinite plate is:

$$K_I = \frac{2}{\pi} \sigma_y \sqrt{\pi a} \qquad \text{(A-8)}$$

where $\sigma$ is the remote stress, and $a$ is the penny shaped crack radius.

As with the 2D case, this solution is modified when an edge crack is considered:

$$K_I = F_I(\varphi) \frac{2}{\pi} \sigma_y \sqrt{\pi a} \qquad \text{A-9)}$$

where $F_I$ is the boundary correction factor, and $\varphi$ is the angle which describes the location of the point of interest on the crack front circumference.

Linear-elastic fracture-mechanics analysis predicts infinite stresses at the crack tip. In real materials, however, stresses at the crack tip are finite because plastic deformation at the crack tip keeps the stress finite. The

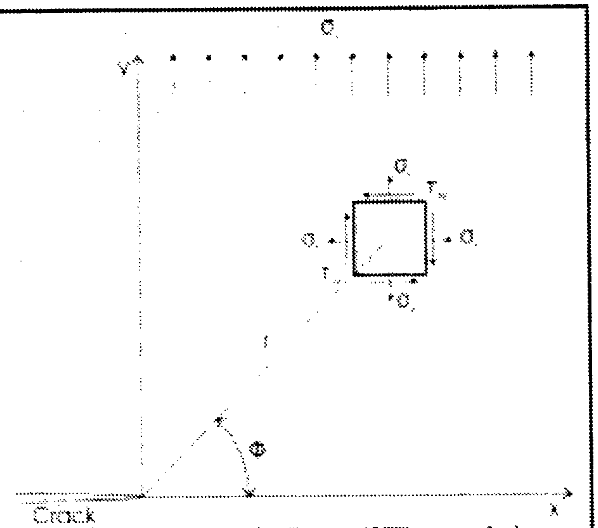


Figure 1 – Stress Intensity Factor (SIF) at crack tip

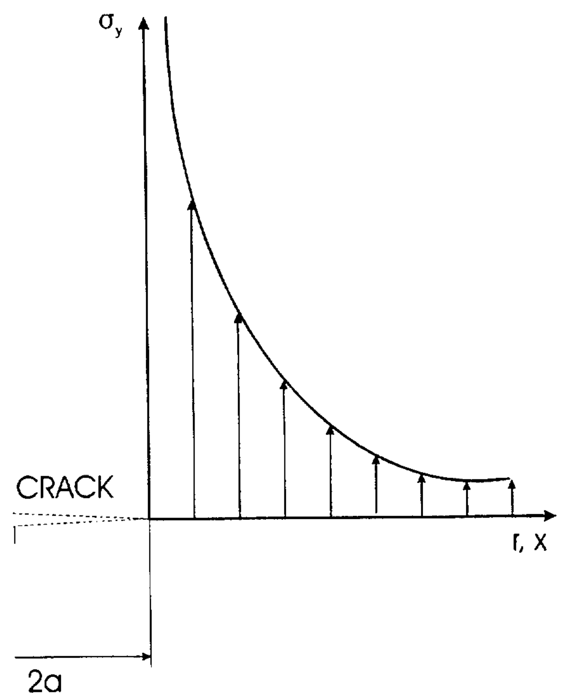


Figure 2 – Elastic stress distribution at crack

simplest estimation of the size of the crack tip plastic zone can be obtained by determining the distance $r_p$ from the crack tip for which the apparent elastic stress $\sigma_y$ is larger than the yield stress, $\sigma_{ys}$, figure 5:

$$r_y = \frac{1}{2\pi} \frac{K^2}{\sigma_{ys}} \quad \text{(for plane stress)} \qquad \text{(A-10)}$$

For conditions of plane strain where the triaxial stress fields suppress the plastic zone size, the plane strain plastic zone size radius is smaller:

$$r_y = \frac{1}{6\pi} \frac{K^2}{\sigma_{ys}} \quad \text{(for plane strain)} \qquad \text{(A-11)}$$

The size of the plastic zone varies along the crack front, being largest at the two free crack surfaces and the

smallest at the mid-plane.

#### 2.2.1.1. *NUMERICAL ANALYSIS OF STRESS INTENSITY FACTOR*

Finite element based stress analysis methods provide displacement and stress distributions in a structure as a primary solution to the problem. If a crack is present in the structure being modelled, post-processing techniques are generally required to extract fracture parameters from these distributions. Three crucial parameters that are extracted from two, and three-dimensional FE calculation results are the stress intensity factor (SIF), the strain energy release rate (SERR) and the J-integral. The most popular techniques for estimating these parameters are:

- Virtual crack extension-based techniques;
- Crack opening displacement-based techniques;
- Stress or nodal-force approximation techniques;
- *J*-integral based techniques.

Because the finite element method is an approximate technique for solving boundary value problems in solid mechanics, each method for extracting SIFs from calculation results has advantages and disadvantages. The choice of a particular method depends on the type of problem analysed. Applications of FEM to fracture mechanics were reviewed, among others, by Pickard (1986), Liebowitz (1989), Banks-Sills (1991).

## 3. PROPERTIES OF GAS TURBINE ENGINE COMPONENT MATERIALS

### 3.1. STATIC PROPERTIES

#### 3.1.1. ENGINEERING STRESS-STRAIN CURVE

An *engineering stress-strain* curve is constructed from *load-elongation* data, in figure 6. The initial portion of this curve OA is the elastic region in which Hooke's law is valid. In this region, stress is proportional to strain. Point A is the elastic limit, defined as the maximum stress that can be applied without creating a permanent strain. The slope of the stress-strain curve in the region OA is the *modulus of elasticity* or *Young's modulus* (*E*). Because the modulus of elasticity is determined by the binding forces between atoms, it is only slightly affected by metal heat treatment and cold work. However, the modulus of elasticity decreases with increasing temperature.

The limit of elastic behaviour of the specimen is described by the *yield strength*, point B at figure 6, defined as the stress which will produce a small amount of permanent deformation, denoted by OC. When the load further exceeds a value corresponding to the yield strength, the specimen undergoes gross plastic deformation, which is demonstrated by permanent deformation of the specimen after the load is released to zero.

As the plastic deformation of the specimen increases, the metal usually becomes stronger due to strain hardening - so that the load required to extend the specimen increases with further straining. Eventually the load reaches the maximum value, point D. The diameter of the specimen decreases as the length increases, since it is usually assumed that total volume of the specimen remains constant during test. From point D further on the diameter of the specimen begins to decrease rapidly because the rate of strain hardening is overtaken by the loss of local bearing area due to necking and internal cavitation. Because of this, the load required to continue deformation drops off until the specimen fractures, Point E in figure 6.

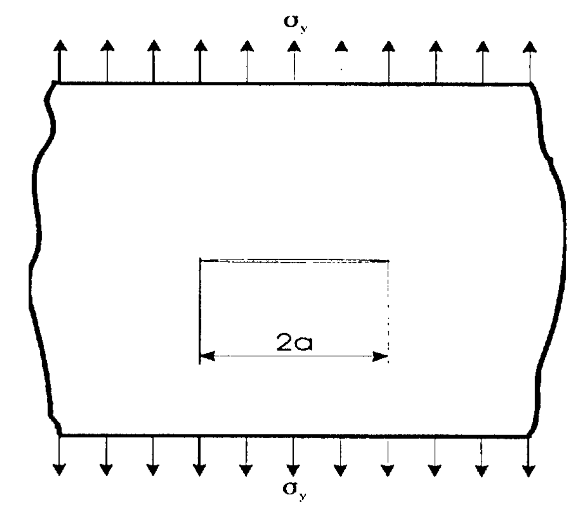



Figure 3 – Crack in infinite plate

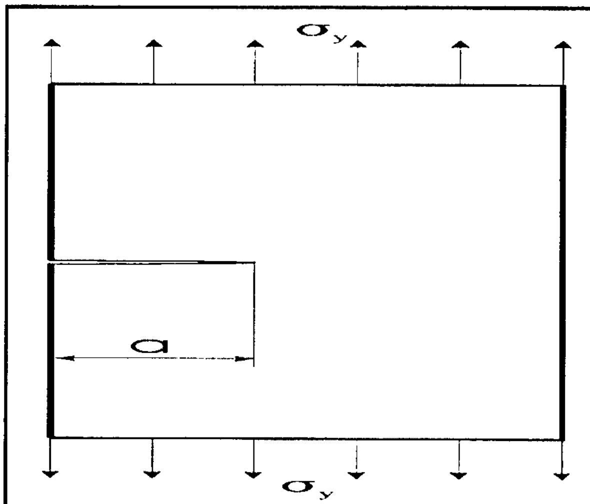



Figure 4 – Edge crack in a half plate

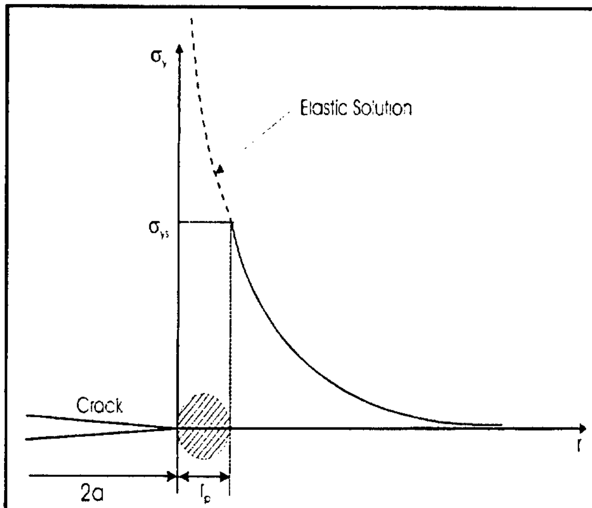



Figure 5 – Plastic zone at crack tip

The point at which plastic deformation begins is hard to define because for most materials there is a gradual transition from elastic to plastic behaviour. Usually the yield strength is defined as the stress required to produce a small specific amount of plastic deformation, so called offset yield strength determined by the value a in figure 6. This offset is usually specified in US practice as a strain of 0.2% or 0.1% (0.002 or 0.001) or 0.1 and 0.5% for UK practice.

If the load is removed at Point B, the unloading follows the path BC. The slope of this unloading curve is parallel to the elastic portion of the curve on loading. The permanent plastic deformation is the offset a in figure 6. However, an elastic deformation is always present in the tension specimen under load.

The tensile strength or ultimate tensile strength (UTS) is the maximum load, Fmax, divided by the original cross-sectional area of the specimen, A0:

$$S_u = \frac{F_{\max}}{A_0} \qquad \text{(A-12)}$$

It should be remembered that although the tensile strength is the value most often quoted from the results of a tension test, it has little fundamental significance with regards to the strength of a metal. This is because the cross al area of the specimen changes continuously as the test progresses.

## 3.2. FLOW PROPERTIES

### 3.2.1. TRUE STRESS-STRAIN CURVE

The *true-stress* σ is expressed in terms of engineering stress *s* by:

$$\sigma = \frac{F}{A_0}(e+1) = s(e+1) \qquad \text{(A-13)}$$

The derivation of this equation assumes constant volume and a homogeneous distribution of strain along the gauge length of the tension specimen. The *true-strain* $\varepsilon$ may be determined from the engineering or conventional strain *e* by:

$$\varepsilon = \ln(e+1) \qquad \text{(A-14)}$$

Both equations are valid until the specimen necking starts. Figure 7 compares the true-stress-true-strain curve with its corresponding engineering stress-strain curve.

In the region of plastic deformation, the flow-curve for metals may be expressed as a simple power law. This is:

$$\sigma = K\varepsilon^n \qquad \text{(A-15)}$$

K is the strength coefficient and n is the strain-hardening exponent. This equation produces a straight line on a double logarithmic curve of the true-stress–true-strain. Typical values for the strain hardening exponent n for metals and alloys range from 0.1 to 0.5. Perfectly plastic solid has n=0, while a perfectly elastic solid has n=1.

The monotonic stress-strain curve can be approximated by the Ramberg-Osgood relationship written in its non-

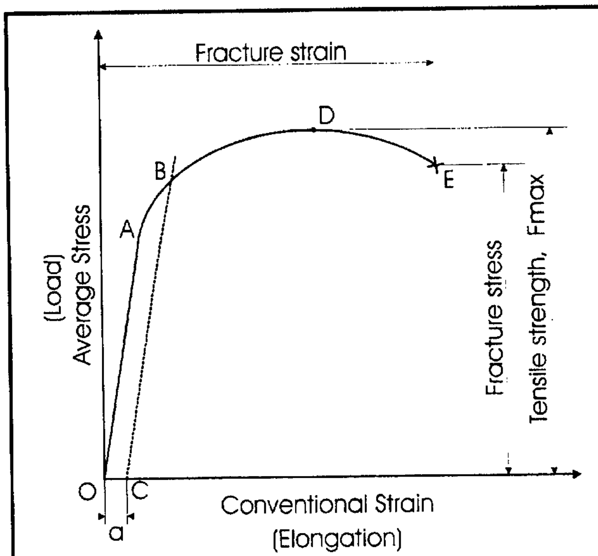



Figure 6 – Engineering stress strain curve

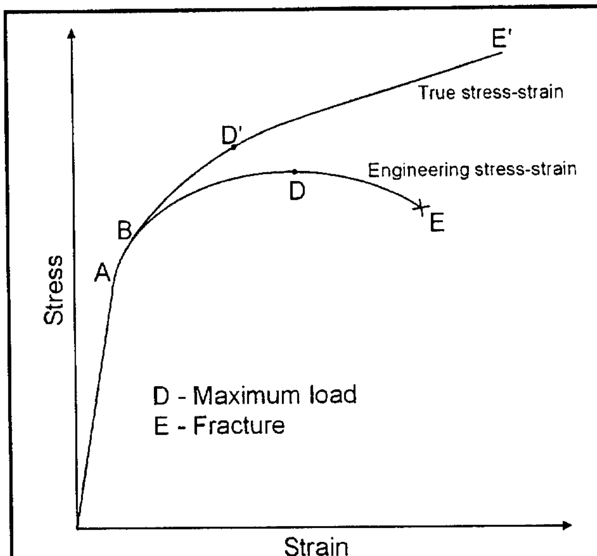



Figure 7 – Limit of elastic behaviour

dimensional form:

$$\frac{\varepsilon}{\varepsilon_0} = \frac{\sigma}{\sigma_0} + \alpha\left(\frac{\sigma}{\sigma_0}\right)^{\frac{1}{n}} \qquad \text{(A-16)}$$

where $\varepsilon_0 = \sigma_0/E$ , $\sigma_0$ is the yield stress and *E* is the Young's modulus of the material.

Relation (A-16) can be rewritten as:

$$\varepsilon = \frac{\sigma}{E} + \alpha\left(\frac{\sigma}{K}\right)^{\frac{1}{n}} \qquad \text{(A-17)}$$

Please note that although the strength coefficient K has the same notation as the stress intensity factor described elsewhere, it represents a different physical quantity.

### 3.2.2. EFFECT OF STRAIN RATE ON FLOW PROPERTIES

A general relation between flow stress and strain rate for constant strain and constant temperature can be expressed

as:

$$\sigma = C\dot{\varepsilon}^m \qquad \text{(A-18)}$$

where $m$ is the strain rate sensitivity coefficient. This relationship is illustrated at figure 7 of chapter 5, which is drawn in a double logarithmic scale.

### 3.2.3. Effect Of Temperature On Flow Properties

The temperature dependence of flow stress at constant strain and strain rate can be generally approximated in the following manner:

$$\sigma = C\,\mathrm{e}^{Q/RT} \qquad \text{(A-19)},$$

where $Q$ is an activation energy for plastic flow, J/(g mole), $R$ is the universal gas constant, J/(K mole), and $T$ is the testing temperature, K.

## 3.3. Creep Properties

The component service temperatures are usually high enough to cause ageing of the material, which results in loss of creep strength and therefore its load bearing capacity.

The shape of a creep curve is determined by several competing reactions, including strain hardening, softening processes such as recovery, recrystalization, precipitate over ageing, damage processes such as cavitation and cracking, and specimen necking. Of these factors, strain hardening tends to decrease the creep rate whereas the other factors tend to increase the creep rate.

Another useful first approximation of the creep strain rate, $\dot{\varepsilon}_c$, as a function of stress, $\sigma$, time, $t$, and temperature, $T$, is to present it as a product of the form:

$$\dot{\varepsilon}_c = f_1(\sigma) f_2(t) f_3(T) \qquad \text{(A-20)}$$

The three functions are defined as below:

(a) The first is a *stress function*, which depends on stress, $\sigma$, only. The most commonly used relations for stress function are:

$$f_1(\sigma) = A_1\sigma^n \qquad \text{(A-21)}$$

$$f_1(\sigma) = A_2 \sinh\left(\frac{\sigma}{\sigma_0}\right) \qquad \text{(A-22)}$$

$$f_1(\sigma) = A_3 \exp\left(\frac{\sigma}{\sigma_0'}\right) \qquad \text{(A-23)}$$

where $A_1$, $A_2$, $A_3$ are constants and $\sigma_0$, $\sigma'_0$ are the reference stresses. The first of these three equations is the most extensively used in practice.

(b)The second is a *time function*: which depends on time, $t$, only It is expressed as a polynomial:

$$f_2(t) = \alpha t^{1/3} + \beta t + \gamma t^3 \qquad \text{(A-24)}$$

where $\alpha$, $\beta$, and $\gamma$ are material constants, which are function of stress and temperature and which relate to primary, secondary and tertiary stages, respectively.

(c) The third is a *temperature function*, which depends on temperature, $T$, only.

In discussion of high-temperature material properties, the temperature is often expressed as a *homologous temperature*. That is the ratio of the test temperature to the melting temperature on an absolute temperature scale. For creep resistant super-alloys, creep becomes significant at a homologous temperature greater than 70%.

The most generally used temperature function is:

$$f_3 = A_0\sigma^n \exp\left[-\frac{Q}{RT}\right] \qquad \text{(A-25)}$$

where $Q$ is the apparent activation energy, $R$ is the universal gas constant and $T$ is the absolute temperature. Both $Q$ and $n$ are material dependent.

It should be remembered that this set of equations is based on the phenomenological approach to creep modelling. This means it is not based on the actual metallurgical processes taking place in the creeping area. Rather it is based on the observed physical behaviour of specimens under test.

Of all the parameters pertaining to the creep curve, the most important for engineering applications are $\varepsilon_r$ and $t_r$. Specifically, their dependence on temperature and applied stress are of extreme importance to the designer. This dependence varies with the applicable creep mechanism.

Damage accumulation during creep can be classified into three categories:

- Damage by loss of internal/external material cross-section;
- Damage by degradation of the micro-structure;
- Damage by environmental attack.

Based on this classification, the continuum damage mechanics, i.e., Kachanov-Rabotnov damage theory, can be used to describe the evolution of creep damage coupled with the creep deformation. However, it does not recognise that inter-granular damage and intra-granular damage occur by different mechanisms and therefore follow different laws. Recent work (Wu and Koul, 1996) has shown that it is necessary to use a super-position strain model, to describe the accumulation of intra-granular damage (e.g. dislocation multiplication and precipitate coarsening), and grain boundary damage (e.g. cavitation and/or oxidation).

In all of the processes described above, diffusion (either grain boundary diffusion or volume diffusion) plays an important role. The diffusion itself can cause deformation under certain conditions. Diffusional flow is generally predominant at stress levels below those of the other mechanisms such as GBS and power-law creep. An example of a deformation mechanism map for the above mechanisms at steady states is shown in figure 8.

## 3.4. Cyclic Properties

The latest approach for some aerospace components is to

classify failures occurring below $10^5$ cycles as LCF and those occurring above $10^7$ as HCF. However, many investigators now define the LCF range to be failure in 50,000–100,000 cycles or less. The area between $10^5$ and $10^7$ is a 'grey area' where attribution of LCF and HCF should performed on a case-by-case basis.

### 3.4.1. Cyclic Stress-Strain Curve

A cyclic stress-strain curve describes stress evolution under strain-controlled cyclic loading. A stress-strain loop under controlled constant strain cycling is presented in figure 9. During initial loading the stress-strain curve is O-A-B. On reloading in tension a hysteresis loop develops. The dimensions of the hysteresis loop are described by its total width $\Delta\varepsilon$, which is the total strain range and $\Delta\sigma$, the total stress range. The total strain range, $\Delta\varepsilon$ consists of the elastic strain component, $\Delta\varepsilon_e=\Delta\sigma/E$, and the plastic strain component $\Delta\varepsilon_p$. Because plastic deformation is not completely reversible, modification of the metal structure occurs during cyclic loading and results in the cyclic stress-strain response.

The cyclic stress-strain curve may be approximated by a power curve similar to the curve for a monotonic stress-strain relationship:

$$\Delta\sigma = K'(\Delta\varepsilon_p)^{n'} \qquad \text{(A-26)}$$

where $K'$ is the cyclic strength coefficient and $n'$ is the cyclic strain hardening coefficient. The equations for the cyclic stress-strain curves are usually presented in the form:

$$\frac{\Delta\varepsilon}{2} = \frac{\Delta\varepsilon_e}{2} + \frac{\Delta\varepsilon_p}{2} \qquad \text{(A-27)}$$

$$\frac{\Delta\varepsilon}{2} = \frac{\Delta\sigma}{2E} + \frac{1}{2}\left[\frac{\Delta\sigma}{K'}\right]^{1/n'} \qquad \text{(A-28)}$$

The cyclic strain hardening exponent, $n'$, has a value in the range of 0.05 to 0.25 for metals and alloys. Cyclically stabilised stress-strain curves provide an important means of characterisation of a material cyclic response. Eq (A-28) is similar to (A-17) except that the strain and stress are replaced by the stress and strain amplitudes and the material constants are denoted by the prime letters.

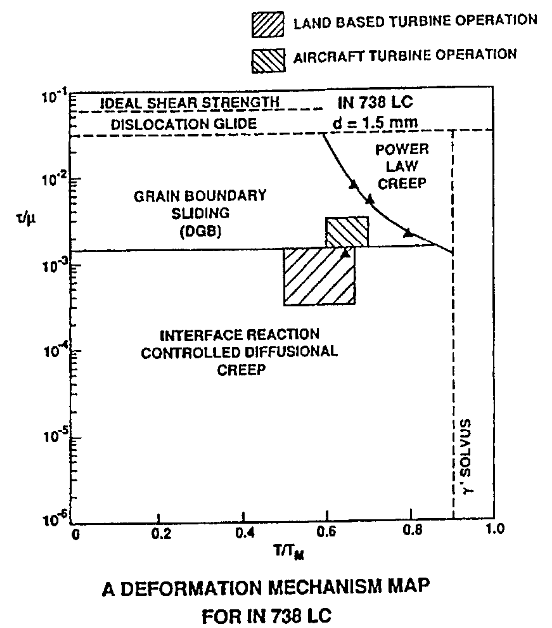



Figure 8 – Deformation mechanisms. After Wu and Koul (1995)

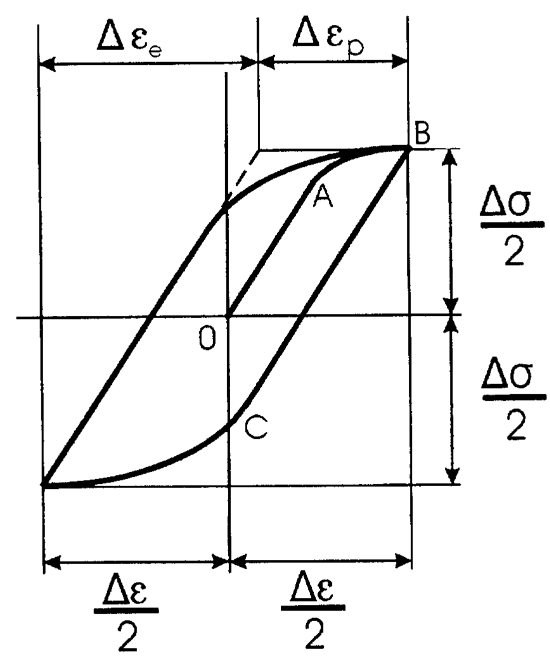



Figure 9 – Cyclic stress-strain loop

### 3.4.2. High Cycle Fatigue (HCF)

The stress ratio is the algebraic ratio of two specified stress values in a stress cycle. Two commonly used stress ratios are the ratio, $A$, of the alternating stress amplitude to the mean stress

$$A = \frac{\sigma_a}{\sigma_m} \qquad \text{(A-29)}$$

and the ratio, $R$, of the minimum stress to the maximum stress:

$$R = \frac{\sigma_{\min}}{\sigma_{\max}} \qquad \text{(A-30)}$$

where the mean stress, $\sigma_m$, is the algebraic average of the maximum and minimum stresses in one cycle:

$$\sigma_m = \frac{\sigma_{\min} + \sigma_{\max}}{2} \qquad \text{(A-31)}$$

and the stress range, $\Delta\sigma$, is the algebraic difference between the maximum and minimum stresses in one cycle:

$$\Delta\sigma = \sigma_{\max} - \sigma_{\min} \qquad \text{(A-32)}$$

High cycle fatigue life assessment is sometimes referred to as the stress-life approach. The basic method for presenting HCF data is by means of S–N curves, which plot the number of cycles to failure, $N$ vs. the stress range. It should be stressed that the S–N curve does not distinguish between crack initiation and crack propagation phenomena.

For some engineering materials, the S–N curve becomes almost flat at low stresses, indicating a threshold value of stress below which failure will not occur for practical purposes. This threshold value of stress, $\sigma_e$, is defined as the fatigue limit.

### 3.4.3. Low Cycle Fatigue (LCF)

Low-cycle fatigue life assessment is sometimes referred to as the strain-life approach. This method is based on the observation that in many components the response of the material in the failure critical locations is strain dependent. In the strain-life approach the plastic strain is directly measured and quantified.

LCF tests are often conducted in the same manner as HCF tests except that the strain range is held constant, and the stress is allowed to vary. The variation of stress with strain in LCF tests typically leads to a hysteresis loop such as the one shown in Figure 9. A tension-compression stress range $\Delta\sigma$ is established corresponding to the strain range $\Delta\varepsilon$ imposed on the specimen. The entire loop width corresponds to the total strain range $\Delta\varepsilon_t$, which can be decomposed into the elastic and plastic strain ranges. The height of the hysteresis loop is the stress range.

Fatigue cracks are mostly initiated at heterogeneous nucleation sites within the material. These could be pre-existing, or generated during the cyclic straining process. Pre-existing nucleation sites can be associated with inclusions, gas pores, or local soft locations in the material microstructure. Fatigue cracks are generated during cyclic straining at the areas where persistent slip bands are being formed in the material.

It has been found that persistent slip bands contain sharp peaks, extrusions, sharp troughs, and intrusions. These result from incomplete slip reversals at the material surface or matrix/inclusion interfaces. These surface or sub-surface notches serve as fatigue-crack nucleation sites. These extrusions and inclusions represent the initial stage in micro-crack formation. It should be recognised that although cracking begins at a surface, the material within these persistent bands and below the surface is also damaged and this will have an influence on the location of the surface crack nucleation. In polycrystaline materials, persistent slip bands can be arrested at grain boundaries, affecting crack nucleation at the junctions of slip bands and grain boundaries.

### 3.4.4. Thermo-Mechanical Fatigue (LCF)

Any model aimed at producing acceptable engine hot-section life-predictions must take into account the component loading cycles, and consider how the stress and inelastic strain at each area of the component vary with time.

Low cycle fatigue results can thus be qualitatively applicable to thermal fatigue, although thermal fatigue problems involve all of the complexities of mechanical loading. In addition, all of the temperature-induced problems are involved. Most low cycle fatigue problems at high temperatures involve thermo-mechanical fatigue processes. Hot section components of gas turbine engines are subjected to cyclic temperature simultaneously with cyclic stress. Analysis of these loads and consideration of the associated fatigue damage become very complex, and many simplifications have to be introduced. Historically, thermal fatigue was considered as isothermal low-cycle-fatigue at the maximum temperature of the thermal cycle to which a component was subjected. However, recent advances in test systems and numerical calculation methods have made it possible to conduct thermo-mechanical fatigue tests under well controlled conditions and also to analyse numerically complex thermo-mechanical cycles. The mechanical low cycle fatigue and thermal low cycle fatigue processes are mathematically expressed by similar set of equations, however using LCF results to predict thermal LCF performance should be undertaken with substantial care. Generally, it is found that TMF loading can be more damaging than pure LCF loading for the same total strain range applied.

Some of the reasons of the differences between thermally and mechanically induced LCF are as follows (Manson, 1966).

Plastic strains in thermal fatigue tend to become concentrated in the hottest region of the body, since the yield point is locally reduced in these locations.

In thermal fatigue, there is often a localised region of strain developed by virtue of plastic flow during the compressive part of the strain cycle to produce a bulging at the hottest region. This is followed by a necking tendency adjacent to the bulge, which is caused during the tensile part of the strain cycle upon cooling, by plastic flow.

Cyclic variation in temperature may have an important effect upon the material properties and it ability to resists LCF failure.

There are interaction effects caused by superposition of simultaneous variations in temperature and strain.

Rates at which the strain cycling is induced may have important effects, since the testing speeds in thermal fatigue tests are often greatly different from the rates used in mechanical LCF tests.

These are compelling reasons to use extreme caution in the prediction of thermal LCF component behaviour from mechanical LCF results and vice versa.

## 3.5. Fracture Mechanics Properties

### 3.5.1. Fatigue Crack Growth Rate (FCGR)

The stress range $\Delta\sigma = \sigma_{max} - \sigma_{min}$ in a given location of a component can be related to the stress intensity factor range, $\Delta K$:

$$\Delta K = K_{\max} - K_{\min} \qquad \text{(A-33)}$$

where $K_{max}$ and $K_{min}$ are the stress intensity factors

associated with the maximum, $\sigma_{max}$, and minimum, $\sigma_{min}$, stresses in a fatigue cycle. This relation, for a particular component, is frequently established using finite element analysis. The growth rate of a fatigue crack as described by Paris or Forman's relation is related to $\Delta K$ and the load ratio, $R$:

$$R = \frac{K_{\min}}{K_{\max}} \qquad \text{(A-34)}$$

which is a ratio of maximal and minimal stress intensity factors in a load cycle.

The Paris formula described by Eq. (6) of chapter 5, is sometimes written as:

$$\frac{da}{dN} = C\left(\frac{\Delta K}{\Delta K_z}\right)^n \qquad \text{(A-35)}$$

where $\Delta K_z$ is the arbitrarily established reference stress intensity factor range. In this equation, the constant $C$ has the unit of m/cycle.

It is necessary to point out that fatigue threshold phenomena described in section 5.5.1 are mostly observed in load shedding FCGR tests using specimens containing a long crack. In these cases, FCGRs are strongly influenced by crack closure associated with the load-shedding procedure (Wu, Wallace and Koul, 1995). The measured $\Delta K_{th}$ value for a long crack in such a test does not represent an intrinsic property of a material. Short crack fatigue experiments show that cracks do grow at $\Delta K$ levels well below the long-crack, $\Delta K_{th}$. Therefore, for the analysis of crack growth near the so-called threshold, $\Delta K_{th}$, it is recommended that short crack data, rather than data obtained for long cracks, is used to assess component lives

Even if the test variables that effect fatigue crack growth rates are stringently controlled, there still remains a significant variation in the results, which must be attributed to the material microstructure. That is, to random distributions of lattice defects, impurity atoms, slip systems, crystal sizes, grain boundary parameters, and macro defects such as porosity and cracks. All of these effects imply the random nature of material damage processes and suggest that a probabilistic approach should be taken when analysing the fracture processes in modern materials. In the probabilistic approach, a random variability of crack sizes is considered as a function of time. This approach requires knowledge of the crack size probability distribution function, the probability distribution function of the extreme loads, and the probability distribution function of the material parameters, such as fracture toughness.

A number of models have been developed to describe the observed variability in crack growth. One of these is the model based on Markov chain theory described by Bogdanoff and Kozin (1985). This model has shown the ability to characterise both constant and variable amplitude fatigue-crack growth. However, this model requires knowledge of the first and second-order moment statistics of time for a crack to reach various sizes.

Another model is obtained by randomising the crack growth rate equation:

$$\frac{da}{dt} = q(a)\,X(t) \qquad \text{(A-36)}$$

where $X(t)$ is the random process, and $q(a)$ is the general, deterministic relation between the crack size $a$ and the crack growth rate. The best results were be obtained if the $X(t)$ process was modelled as a random pulse train, (Lin and Yang, 1983). Other aspects of the statistical modelling of the FCGR can be found in Besuner (1987).

### 3.5.2. Creep Crack Growth Rate

The most commonly used parameters for describing creep crack growth rate $\dot{a}$ are stress intensity factor $K$, and the creep fracture mechanics term $C^*$. The following relations have been produced, mostly to correlate individual test data:

$$\dot{a} = AK^m \qquad \text{(A-37)}$$

$$\dot{a} = D_0 C^{*\phi} \qquad \text{(A-38)}$$

where $A$, $Do$, $m$, and $\phi$ are material constants which may be temperature and stress state dependent. The creep fracture mechanics term $C^*$ is the contour integral characterising steady state creep. It is similar to J-integral, which characterises the fatigue crack growth rate in elastic-plastic fracture. The exponent $m$ has a value close to the value of n in Eq. (A-25), while the exponent $\phi$ is a fraction close to unity (Webster and Ainsworth, 1994).

## 3.6. Influence of Mean Stress on Fatigue and Fracture

### 3.6.1. Influence of Mean Stress on HCF

The curves, which show the dependence of limiting range stress, $\sigma_{max}$-$\sigma_{min}$, on mean stress are called Goodman diagrams. An example of which is shown in figure 10. This relationship is established for a fixed number of cycles or at the fatigue limit. It can be seen in figure 10, that as the mean stress becomes more tensile, that the allowable stress range reduces. A conservative approximation of a Goodman diagram may be obtained by drawing straight lines from the fatigue limit for completely reversed cycles to the tensile strength.

The other method of presenting the mean stress effect on HCF is known as a Haig-Soderberg diagram in which the alternating stress is plotted against the mean stress, figure 11. These plots were historically the subjects of numerous empirical curve-fitting procedures. A straight line represents the Goodman method while the parabolic curve was suggested by Gerber. If the design is based on the yield strength, as shown by a dashed line in figure 10, than $\sigma_0$ is substituted for $\sigma_u$.

The equations for these three relationships can be presented as follows:

Goodman's Linear Relation

$$\frac{\sigma_a}{\sigma_N}+\frac{\sigma_m}{\sigma_u}=1 \qquad \text{(A-39)}$$

Gerber's Parabolic Relation

$$\frac{\sigma_a}{\sigma_N}+\left(\frac{\sigma_m}{\sigma_u}\right)^2=1 \qquad \text{(A-40)}$$

Soderberg's Linear Relationship

$$\frac{\sigma_a}{\sigma_N}+\frac{\sigma_m}{\sigma_{yp}}=1 \qquad \text{(A-41)}$$

where $\sigma_a$ is the alternating stress amplitude, $\sigma_m$ is the mean stress, $\sigma_N$ is the fatigue strength at $N$ cycles and, $\sigma_{yp}$ is the yield point of the material.

### 3.6.2. Influence of Mean Stress on LCF

Walker (1970) proposed a method of taking stress ratio into account during crack propagation and fatigue in the form of an equation:

$$\frac{da}{dN}=C\left[(1-R)^m K_{\max}\right]^n \qquad \text{(A-42)}$$

where $R$ is the stress ratio, $C$ and $n$ are the appropriate parameters of the Paris law, Eq(6) of chapter 5, and $m$ is the experimentally established Walker constant having a value between 0 and 1.

This Walker method used for correction for non-zero mean stress in the FCGR analysis is also applied, as a way of deriving material LCF data for stress conditions other than zero minimum stress ($A \neq 1$), Eq (8) of chapter 5.

With respect to the Walker model, it should be remembered that:

- Not all Walker parameters are applicable to all materials and to all stress ratios: each material and stress ratio is usually defined by a unique Walker parameter.
- The method is most effective in $R$ ratio range of −1 to 0.6 ($A$ ratio from infinity to $0.25$).
- It is a compromise in modelling behaviours at various mean stresses, $A$ or $R$ ratios; there may be an uneven compromise at one or more sets of $A$-ratio data.
- It can lead to unrealistic trends when very low and very high stress ratios are involved.
- The mean stress effect is more pronounced at room temperature and drops off as temperature increases. Therefore, the Walker exponent is a function of temperature.
- Significant compressive stress or negative mean stress invalidates the method.

### 3.7. Influence of Multiaxial State of Stress on Fatigue and Fracture

Combined fatigue under high-strain low-cycle conditions can be correlated with the octahedral shear strain:

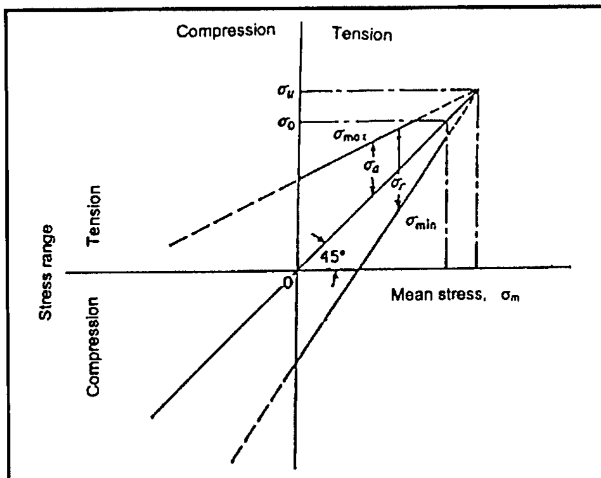



Figure 10 – Goodman diagram. After Dieter (1986)

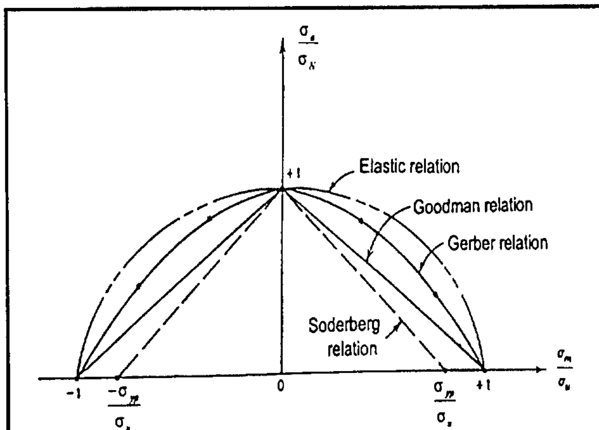



Figure 11 – Haig-Soderberg diagram. After Collins (1993)

$$\varepsilon_q=\frac{\gamma_{oct}}{2}=\frac{1}{3}\left[(\varepsilon_1-\varepsilon_2)^2+(\varepsilon_2-\varepsilon_3)^2+(\varepsilon_3-\varepsilon_1)^2\right]^{\frac{1}{2}} \qquad \text{(A-43)}$$

where $\varepsilon_1$, $\varepsilon_2$, $\varepsilon_3$ are the principal components of strain, and $\gamma_{oct}$ is the octahedral strain.

The combined strain-state is consolidated in this equation to provide $\varepsilon_q$, which is entered into the Coffin-Manson formula, Eq (3).

Similarly, it was proposed to use the following expression for fatigue failure under combined stress in low strain conditions – high cycle fatigue:

$$\left[(\sigma_{a1}-\sigma_{a2})^2+(\sigma_{a2}-\sigma_{a3})^2+(\sigma_{a3}-\sigma_{a1})^2\right]^{1/2}+ C_2(\sigma_{m1}+\sigma_{m2}+\sigma_{m3})\geq\frac{\sqrt{2}\sigma_e}{K_f} \qquad \text{(A-44)}$$

where

- $\sigma_{ai}$ is the alternating component of principal stress in the '1' direction;
- $\sigma_{mi}$ is the static component of principal stress in '1' direction;

- $\sigma_e$ is the fatigue strength (fatigue limit) for completely reversed stress;
- $K_f$ is the fatigue notch factor, i.e the ratio of the fatigue limit of unnotched specimens to the fatigue limit of notched specimens,
- C2 is the material constant taking into account the influence of $\sigma_m$ on $\sigma_a$. At first approximation $C_2$ = 0.5.

If fatigue properties are anisotropic, or become anisotropic due to the effect of cyclic plasticity on material properties, then the concept of a single equivalent strain or stress is no longer valid. This is because the direction of loading with respect to the direction of fatigue properties may become an important factor.

Many questions remain to be answered with respect to the validity of the above equations (Laflen and Cook, 1982).

# 4. ANALYTICAL TECHNIQUES FOR DAMAGE CHARACTERISATION

## 4.1. CUMULATIVE FATIGUE DAMAGE

### 4.1.1.1. DOUBLE LINEAR DAMAGE RULE

It was proposed that cumulative damage estimates might be improved by breaking the fatigue process down into a crack initiation phase and a crack propagation phase and applying the linear damage rule to each phase separately. Manson (1967) suggested that a crack propagation period can be modelled as

$$N_p = PN_f^p \qquad \text{(A-45)}$$

where,

- $N_p$ is the number of cycles to propagate a crack to failure after it has been initiated,
- $N_f$ is the total number of cycles to failure,
- $P$ is the experimentally determined propagation coefficient,
- $p$ is the experimentally determined propagation exponent.

In the original work, the coefficients $P$ and $p$ were experimentally established as 14 and 0.6, respectively.

Crack initiation period can just be obtained by subtracting the number of cycles to propagate the crack from the total number of cycles:

$$N' = N_f - N_p \qquad \text{(A-46)}$$

A linear damage rule is applied to each phase individually to produce predictions of crack initiation and component failure as follows:

Fatigue cracks of critical size are initiated when

$$\sum_{i=1}^{m} \frac{n_i}{N'_i} = 1 \qquad \text{(A-47)}$$

Fatigue crack are propagated to failure when:

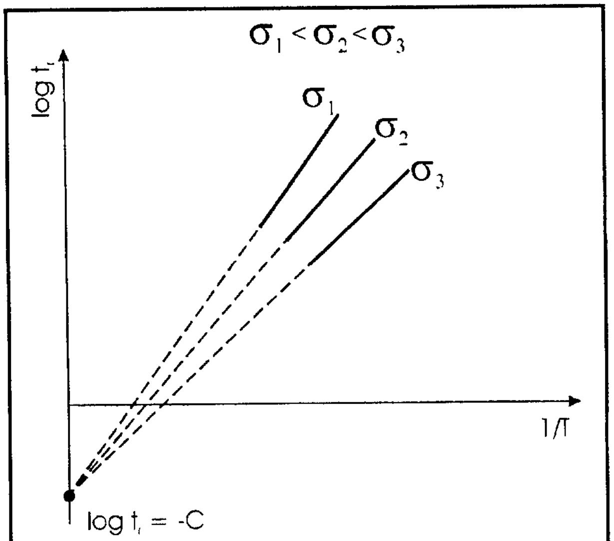


Figure 12 – Larsen-Miller time temperature grouping

$$\sum_{i=1}^{m} \frac{n_j}{(N_p)_j} = 1 \qquad \text{(A-48)}$$

In each case $n$ is the number of cycles applied at the i[th] and j[th] stress levels.

The double linear damage rule gave good agreement with test data for two stress level tests for several materials.

## 4.2. CREEP DAMAGE

### 4.2.1. LARSON-MILLER PARAMETERS

A concept of a time-temperature grouping in the form $T(K_1 + \log t)$ was introduced by Larson and Miller (1952). For a given material a plot of stress vs. the above parameter resulted in a single plot regardless of the time-temperature combination employed to derive the parameters, such as shown in figure 12. It can be shown that the Larson-Miller parameter can be readily derived from the stress and temperature dependence of the creep rate at time to rupture. The rate equations can generally be written as:

$$t_r = A_2 \exp(\frac{B_2}{T}) \qquad \text{(A-49)}$$

$$\dot{\varepsilon} = A_1 \exp(-\frac{B_1}{T}) \qquad \text{(A-50)}$$

This parameter can be expressed in terms of time to a specific strain or time to rupture. This relation implies that the activation energy is dependent on stress. The Larson-Miller parameter is usually written in the form:

$$P_t = f(\sigma) = T(C + \log_{10} t) \qquad \text{(A-51)}$$

where P is the Larson-Miller parameter which is a function of stress only, $C$ is a constant and t is the time to a specific strain or time to rupture. A plot of $\log t_r$ vs $1/T$

results in straight lines whose intercept is a constant and whose slope is a function of stress as shown in figure 12.

### 4.2.2. Monkman-Grant Type Relationship

The Monkman-Grant relationship is effectively a critical strain criterion. It states that the strain accumulated during secondary creep is a constant at failure so that the product of the secondary creep rate and the rupture life $t_r$ is constant:

$$\dot{\varepsilon}\, t_r = C_{MG} \tag{A-52}$$

where CMG is the Monkman-Grant constant. When secondary creep is dominant, constant creep ductility independent of stress and temperature is predicted.

### 4.2.3. Theta Projection Concept

A very general description of the creep curve under constant-stress conditions is given by the $\theta$ projection concept in which creep strain, $\varepsilon$, is considered to be the sum of two competing processes according to the equation:

$$\varepsilon = \theta_1 [1 - \exp(-\theta_2 t)] + \theta_3 [\exp(\theta_4 t) - 1] \tag{A-53}$$

In this expression, $\theta_1$, $\theta_2$, $\theta_3$, and $\theta_4$ are all experimentally determined constants which are functions of stress and temperature. $\theta_1$ and $\theta_2$ define the primary or decaying strain-rate component, and $\theta_3$, and $\theta_4$ describe the tertiary or accelerating strain rate component. A wide range of creep curve shapes can be modelled with various combinations of these constants.

## 4.3. Creep-Fatigue Interaction

### 4.3.1. Linear Damage Summation

The most common approach is based on the linear superposition of fatigue and creep damage. In this approach it is assumed that the damage fraction occurring in fatigue and the fatigue damage fraction occurring in creep may simply be added. This approach combines the damage summation of Miner's rule and Robinson's creep damage rule as follows:

$$D' = \sum \frac{N}{N_f} + \sum \frac{t}{t_r} \leq 1 \tag{A-54}$$

where $N/N_f$ is the cyclic portion of the life fraction, in which $N$ is the number of cycles at a given strain range and $N_f$ is the pure fatigue life at that strain range. The time dependent creep-fatigue fraction is $t/t_r$, where t is the time at a given stress and $t_r$ is the time to rupture at this stress. The stress relaxation period is divided into time blocks during which an average, constant value of stress prevails, and for each time block $t/t_r$ is computed and summed. $D'$ is the cumulative damage index, such that when $D \fallingdotseq 1$, failure is presumed to occur.

Unfortunately, several material and test parameters may affect the distribution of $D'$, and there is no satisfactory way of applying the linear damage rules at present. A serious drawback of this approach is that the life fraction rule is purely phenomenological, having no mechanistic basis. Its applicability is therefore material dependent. Contrary to experience, it also assumes that tensile and compressive dwell periods are equally damaging. This approach is sensitive to loading conditions and dwell time effects although it is consistent for varying temperature conditions. More importantly though, it has been shown that significant interaction, which reduces the overall life of the component, can occur.

Specimen testing shows that a reduction in life can be obtained with a relatively small percentage of mixed creep and fatigue. Loading histories and temperatures should be evaluated to establish which of these effects are likely to occur. From a practical point of view, when low combined creep-fatigue lives are calculated, detailed specimen and component testing is performed to validate the calculations. Despite these limitations, the damage summation method is very popular because it is easy to use and requires only standard S-N curves and stress rupture curves.

### 4.3.2. Strain Range Partitioning

The strain range partitioning (SRP) approach involves partitioning of the total inelastic strain range into four possible components depending on:

- Whether the stresses are tensile or compressive;
- The type of inelastic strain accumulated (creep or time independent plasticity).

Figure 13 shows a typical stress-strain hysteresis loop, and figure 14 shows the four generic types of hysteresis loops for the four types of strain range. The hysteresis loop from a creep-fatigue test (i.e. LCF test with dwell time) is broken down into the component strains: $\Delta\varepsilon_{pp}$, $\Delta\varepsilon_{cc}$, $\Delta\varepsilon_{pc}$, $\Delta\varepsilon_{cp}$. The terms $\Delta\varepsilon_{pp}$ and $\Delta\varepsilon_{cc}$ represent the pure reversed plastic and reversed creep ranges, respectively and the two other terms represent combined creep and plastic strain ranges. For each type of strain range, the Coffin-Manson relationship, Eq. (A-3) can be applied. For instance:

$$N_{pp} = A(\Delta\varepsilon_{pp})^{\alpha} \tag{A-55}$$

The fractional strain for each type of strain with respect to the total inelastic strain is expressed, for example, as:

$$F_{pp} = \frac{\Delta\varepsilon_{pp}}{\Delta\varepsilon_{in}} \tag{A-56}$$

By adding up the fractional damage for each type of strain, the total damage is estimated by the expression:

$$\frac{1}{N_f} = \frac{F_{pp}}{N_{pp}} + \frac{F_{cc}}{N_{cc}} + \frac{F_{pc}}{N_{pc}} + \frac{F_{cp}}{N_{cp}} \tag{A-57}$$

where $N_{pp}$, $N_{cc}$, etc. represent the number of cycles to failure for each type of strain.

One of the major problems with this approach is the need to generate baseline data based on complex dwell-period tests. Extrapolation of predictions to long dwell periods and small strain ranges also needs further verification. In

order to apply Eq. (A-57) to life prediction for any arbitrary cycle, the following information is needed:

a) From a stable hysteresis loop of the stress-strain cycle, the partitioned strain range components $\Delta\varepsilon_{pp}$, $\Delta\varepsilon_{cc}$, $\Delta\varepsilon_{pc}$ *(or* $\Delta\varepsilon_{cp}$) and the total inelastic strain range $\Delta\varepsilon$ are obtained.

b) The fractional strains $F_{pp}$, $F_{cc}$ and $F_{cp}$ are then calculated by use of the information obtained in step 1 and by use of Eq (A-56).

c) The number of cycles to failure for each given type of strain (i.e the relationships $N_{pp} = A\ \Delta\varepsilon_{pp}$, etc) must be known from independent laboratory experiments.

d) Now that the fractional strains $F_{pp}$, $F_{cc}$, etc. and the cyclic life for each type of strain, $N_{pp}$, $N_{cc}$, etc. are known, the interaction damage rule Eq (A-57) is used to calculate the cyclic life for the arbitrary cycle.

### 4.3.3. Ductility Exhaustion

The ductility exhaustion approach is simply a strain-based life prediction rule in which the fatigue damage and creep damage are summed in terms of the fractional strain damage for each category, as follows:

$$\frac{1}{N_f} = \frac{\Delta\varepsilon_p}{D_p} + \frac{\Delta\varepsilon_c}{D_c} \qquad \text{(A-58)}$$

where $\Delta\varepsilon_p$ is the plastic strain-range component at half life, $D_p$ is the fatigue ductility obtained from pure fatigue tests, $\Delta\varepsilon_c$ is the true tensile creep-strain component, and $D_c$ is the lower-bound creep-rupture ductility of the material. The first term in this equation denotes the fatigue-damage component, and the second term denotes the creep-damage components. The ductility-exhaustion approach is simple to use and has some mechanistic basis. Selection of appropriate values for $D_c$ and $\Delta\varepsilon_c$ is, however, arbitrary and subject to errors.

## 5. References

See Chapter 5.

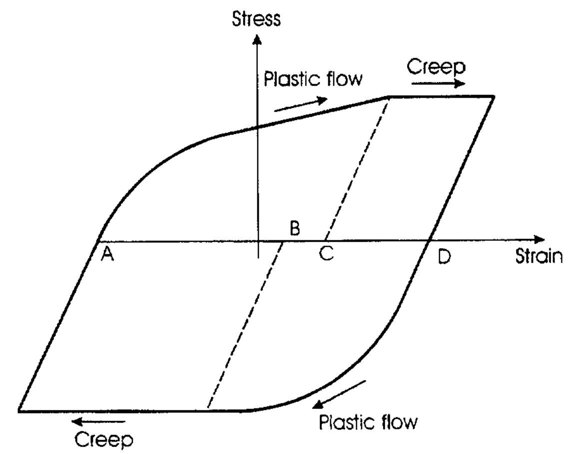



Figure 13 – Stress-strain hysteresis loop

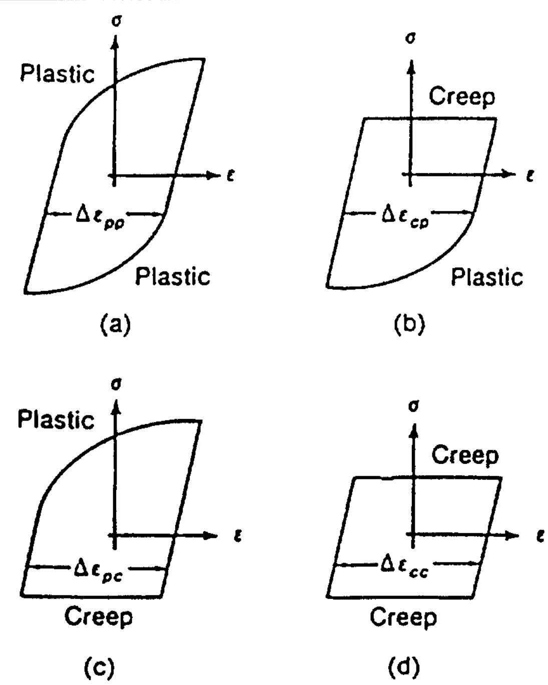



Figure 14 – Four generic types of hysteresis loop

# Glossary

| Term | Meaning |
|---|---|
| Accelerated Mission Testing (AMT) | Testing programme designed to expose selected types of design defect on one or more test-bed engines, or models. |
| Accelerated Simulated Mission Endurance Testing (ASMET) | As AMT. |
| Airworthiness | Fitness to fly. |
| Availability | The percentage of a fleet of equipment that is available for use. |
| Banding | A method used for assigning usage damage to components. The bands are typically engine speed ranges. Also known as gating. |
| Calibration | Comparison and recording of the differences between a known standard and an instrument that measures it. |
| Component | Part of an engine. An item that forms part of an assembly. |
| Condition monitoring | Monitoring the ability of an equipment or system to perform its function. |
| Creep | Permanent deformation that occurs as a result of the prolonged application of stress. The applied stress is below a level that would be expected to cause plastic deformation. Occurs at temperatures that are high relative to the melting point of the material. |
| Creep, *Mechanical* | Permanent elongation of a component when exposed at high temperatures to stresses that would only cause elastic deformation at low temperature. As above. |
| Creep, *Metallurgical* | Very local yielding of material in stressed components at high temperatures. The effect is to redistribute local stresses, without any apparent gross deformation of the component. |
| Cumulative damage | The sum of damage which occurs over a period. |
| Damage | Physical changes in a component, which cause it to deviate from its 'as new' condition. Some forms of damage, such as fatigue, may not be visible to the naked eye. |
| Damage tolerance | A design philosophy for safety critical parts. The basic assumption is that some form of metallurgical damage exists in all new metallic components. It is assumed that a finite time will be required for the damage to become a detectable defect. Design techniques that assume that such damage exists, and allow the component to be used safely until the damage increases to a pre-determined limit are known as 'damage tolerance' design techniques. |
| Data | Records of observations. Used in both the singular and the plural. |
| Duration | A period (of time). |
| Engine | In the context of this document, a gas turbine engine. |
| Engine flight time | Length of period when engine is airborne. |
| Engine module | A designated and interchangeable engine sub assembly. In many cases, it may have the same traceability requirements as a complete engine. |
| Engine Life Monitoring System (ELMS) | The complete monitoring system including all airborne and ground based equipment. |
| Engine Life Monitoring Unit (ELU or EMU) | An airborne unit which processes and records relevant engine and aircraft data. |
| Engine qualification testing | Formal tests that all new engine types must pass to demonstrate that the design and construction meets the required airworthiness regulations and standards. |
| Engine running time | Total recorded engine-running time. In some cases, this may not include test-bed and on-wing maintenance running. |
| ENSIP | Engine Structural Integrity Program. Used for proving the safety of US military engines. |
| Equipment | An item designated as 'equipment' for ease of identification and logistical purposes. |
| Estimate | A value that is calculated, or indirectly determined. Not measured. |
| Exchange rate | A number used to convert flying hours into equivalent usage or damage counts. Also known as Beta factor. |
| Fatigue, *Low Cycle* | Damage that is related to stress cycles that lead to component failure in less than $10^5$ or $10^7$ stress cycles. Stress and temperature cycles are of high magnitude at relatively low frequency causing large high plastic strain over a microscopic region. Normal design practices and calculations consider this type of damage. It is the major life limitation on many rotating components, shafts and combustion chamber outer casings. |
| Fatigue, *High Cycle* | Damage that is related to stress cycles that lead to component failure in more than $10^7$ stress cycles, due primarily to vibration. Occurs at stress cycles of low magnitude at high frequency causing elastic microscopic strain or very small amounts of plastic yielding. This |

| | |
|---|---|
| | type of damage is usually avoided through the development testing process, and detail design practices rather than by detailed design calculation. Many HCF failures are initiated by other damage, such as LCF cracks. |
| Fatigue, *Thermal* | Damage that is induced by thermally induced stresses. |
| Flight | The period from the wheels leaving the ground, until they touch again. Many definitions exist. In some cases, a number of "Touch and go" manoeuvres would be counted as a single flight. This is an operational convenience. In component lifing terms, the accumulated damage on such a flight may be much higher than that on a normal flight. |
| Fracture critical part | A part which will physically break, causing catastrophic damage, after experiencing a statistically described number and mix of missions. Such components are identified at design time, and removed from service before failure occurs. |
| Functional failure | When an equipment no longer performs as required, or as designed. |
| Information Management System (IMS) | A system which stores data and provides an interrogation and analysis facility. Part of an engine life monitoring system. |
| Life count | A value related to the number of design reference cycles that a component is designed to endure before failure. It is estimated or measured by a usage monitoring system. |
| Life-critical part | A part which limits the in-service life of an engine or module. |
| Life, *dysfunction* | The life at which a part is judged to have failed. This is based on a combination of the defect initiation life and the subsequent crack propagation life. 2/3 of the dysfunction-life may be taken as a statistical adjustment to allow for variations in materials properties, and applied as the service life. |
| Life expiry | Consumption of all of the permitted life counts. |
| Life extension | An increase in the permitted number of life counts. Fracture mechanics techniques or other methods may be employed to show that the life increase is safe. |
| Life, *Safe* | The formally declared number of life counts or flying hours that a component may accumulate in use. |
| Life usage | The life usage counts or hours accumulated by a component. |
| Lifed component | A component that has a formally declared design life. |
| Lifing | Assigning a life in hours or reference stress cycles to a component. |
| Lift-off | The moment when the wheels of an aircraft leave the ground. |
| Maintenance, *On-Condition* | Maintenance actions that are taken only if signs appear, which show that a loss of operational functionality is likely to occur. |
| Maintenance management | The art or science of organising maintenance procedures and practice. |
| Maintenance, *Preventive* | Maintenance done before any loss of performance, to protect future operational functionality. |
| Maintenance, *Reliability Centred* | Process that determines what must be done to ensure that any asset continues to fulfil its intended function in its operating context. |
| Maintenance, *Scheduled* | Maintenance done according to a time schedule, regardless of actual need. |
| Maintenance, *Unscheduled* | Maintenance done between scheduled maintenance periods, to restore lost functionality. |
| Mission | A military operation. Also known as sortie |
| Mission mix | A weighted mixture of mission profiles used to estimate the likely life usage in service. |
| Mission profile | A definition of a sequence of flight manoeuvres for a particular type of mission. |
| Mission profile code | An identifying code assigned to each of the types of mission flown by a particular aircraft type. Also known as sortie-pattern code. |
| Mission severity factor | An airframe fatigue damage indicator assigned to each mission profile code. |
| Model | A physical, software or theoretical representation of some reality. |
| Module | An engine assembly that is designated as a module by the manufacturer. Modules usually have log cards. |
| Monitoring | Measuring, recording and making decisions based on the recorded data. |
| Operational | Relating to service use. |
| Partitioning | Separating different types of failure mechanism, such as strain-range. |
| Parts life tracking | Recording the life consumption of each lifed item on an individual basis. |
| Plausibility test | A test which data must pass before it is accepted for use. |
| Reference cycle | A stress-strain cycle for a stress feature. Derived from the mission mix, and used by the designer as the basis for calculating damage and consumed life counts. |
| Reliability | The probability of failure of an equipment or component, in a period. Normally expressed as mean time between failures. Critical components are not supposed to fail in service. |

| | |
|---|---|
| | Therefore, their reliability cannot be proven and expressed in the same way. |
| Rainflow analysis | A method of extracting usage cycles from a time history of turning points. The greatest maximum is matched to the lowest minimum, considered as a cycle and extracted from the time history. This process is repeated until all cycles have been identified. |
| Safety | A measure of operational success, in terms of deaths or injuries per some operating period. |
| Safety critical component | A component, the failure of which, endangers the safety of the aircraft or people in the vicinity. |
| Snap-shot | Collection of data at a defined point in the flight cycle. |
| Strain | Deformation. |
| Strain, *Elastic* | Temporary deformation that recovers when the load is removed. |
| Strain, *In-elastic* | Permanent deformation. |
| Strain, *Mechanical* | Overall deformation or elongation of a component, relative to its original length. |
| Strain, *True* | Point or incremental deformation of material relative to the instantaneous incremental length. |
| Stress | A load applied to an area of material. Stresses are normally resolved into two components - Normal and shear. They apply to all three axes. |
| Stress feature | A part of a component that may be the life limiting feature under some conditions. Some components have several stress features which are the subject of intensive design effort. |
| Stress rupture | Failure when a component has been statically loaded at an elevated temperature for a long period. |
| Stress, *Normal* | The stress perpendicular to a surface of an elemental cube. |
| Stress, *Shear* | The stresses lying in the plane of a surface of an elemental cube. |
| Thermal transient | When the surface temperature of a component changes it takes some time for the temperature distribution across the component to stabilise. The presence of thermal gradients induces stresses. As the gradient changes, so do the stresses. |
| Thermo-mechanical behaviour | The materials data used to make life assessments, is isothermal data implying that the strain is cycled at constant temperature. In reality, both the strain and temperature will vary throughout the cycle and therefore consideration of the thermo-mechanical fatigue behaviour is necessary. Large test programs are underway to understand the differences between TMF and isothermal data. Early evidence suggests that in certain situations an over-prediction of life is obtained and that in other areas an under-prediction is predicted when using isothermal data. |
| Touchdown | The moment when the wheels of an aircraft touch the ground. |
| Trend analysis | A variation of rain-flow analysis, which seeks to reduce the complexity of the algorithm with negligible effect on the accuracy of the result. |
| Type test | A formal test-bed engine test that proves the basic safety of an engine prior to release into service. |
| Usage monitoring | Estimating and recording of the usage of a component. Frequently used to refer to life monitoring. |
| Verification | Verification is proving the truth of an element of a system. |
| Validation | Validation is proving the truth of a complete system. |

# REPORT DOCUMENTATION PAGE

| 1. Recipient's Reference | 2. Originator's References | 3. Further Reference | 4. Security Classification of Document |
|---|---|---|---|
| | RTO-TR-28<br>AC/323(AVT)TP/22 | ISBN 92-837-1032-0 | UNCLASSIFIED/<br>UNLIMITED |

**5. Originator** Research and Technology Organization
North Atlantic Treaty Organization
BP 25, 7 rue Ancelle, F-92201 Neuilly-sur-Seine Cedex, France

**6. Title**
Recommended Practices for Monitoring Gas Turbine Engine Life Consumption

**7. Presented at/sponsored by**
the RTO Applied Vehicle Technology Panel (AVT).

| 8. Author(s)/Editor(s) | 9. Date |
|---|---|
| Multiple | April 2000 |

| 10. Author's/Editor's Address | 11. Pages |
|---|---|
| Multiple | 178 |

**12. Distribution Statement** There are no restrictions on the distribution of this document. Information about the availability of this and other RTO unclassified publications is given on the back cover.

**13. Keywords/Descriptors**

Equipment health monitoring
Machinery health monitoring
Aircraft maintenance
Service life
Life cycle costs
Gas turbine engines
Operations
Military aircraft
Commercial aircraft
Aging (metallurgy)
Damage
Tolerances (mechanics)
Gas turbine rotors
Safety
Turbine components
Failure
Control equipment
Standards
Structural integrity
Fracture mechanics
Fatigue (materials)
Aging aircraft

**14. Abstract**

The Task Group analysed the use of life monitoring systems in modern engines (from 1990) and in ageing fleets. The design and operational factors to be considered beforehand are described.

Particular attention is paid to turbine disks. Regulatory requirements for safety standards are considered. Civil military practices, maintenance policies and procedures, modes and mechanics of service usage are covered as well as their influence on life consumption. Lifing procedures, monitoring system verification and validation, operational management considerations and usage monitoring approaches are dealt with.

NORTH ATLANTIC TREATY ORGANIZATION

RESEARCH AND TECHNOLOGY ORGANIZATION
BP 25 • 7 RUE ANCELLE
F-92201 NEUILLY-SUR-SEINE CEDEX • FRANCE
Télécopie 0(1)55.61.22.99 • E-mail mailbox@rta.nato.int

**DIFFUSION DES PUBLICATIONS**
**RTO NON CLASSIFIEES**

L'Organisation pour la recherche et la technologie de l'OTAN (RTO), détient un stock limité de certaines de ses publications récentes, ainsi que de celles de l'ancien AGARD (Groupe consultatif pour la recherche et les réalisations aérospatiales de l'OTAN). Celles-ci pourront éventuellement être obtenues sous forme de copie papier. Pour de plus amples renseignements concernant l'achat de ces ouvrages, adressez-vous par lettre ou par télécopie à l'adresse indiquée ci-dessus. Veuillez ne pas téléphoner.

Des exemplaires supplémentaires peuvent parfois être obtenus auprès des centres nationaux de distribution indiqués ci-dessous. Si vous souhaitez recevoir toutes les publications de la RTO, ou simplement celles qui concernent certains Panels, vous pouvez demander d'être inclus sur la liste d'envoi de l'un de ces centres.

Les publications de la RTO et de l'AGARD sont en vente auprès des agences de vente indiquées ci-dessous, sous forme de photocopie ou de microfiche. Certains originaux peuvent également être obtenus auprès de CASI.

## CENTRES DE DIFFUSION NATIONAUX

**ALLEMAGNE**
Streitkräfteamt / Abteilung III
Fachinformationszentrum der Bunderswehr, (FIZBw)
Friedrich-Ebert-Allee 34
D-53113 Bonn

**BELGIQUE**
Coordinateur RTO - VSL/RTO
Etat-Major de la Force Aérienne
Quartier Reine Elisabeth
Rue d'Evère, B-1140 Bruxelles

**CANADA**
Directeur - Recherche et développement - Communications et gestion de l'information - DRDCGI 3
Ministère de la Défense nationale
Ottawa, Ontario K1A 0K2

**DANEMARK**
Danish Defence Research Establishment
Ryvangs Allé 1, P.O. Box 2715
DK-2100 Copenhagen Ø

**ESPAGNE**
INTA (RTO/AGARD Publications)
Carretera de Torrejón a Ajalvir, Pk.4
28850 Torrejón de Ardoz - Madrid

**ETATS-UNIS**
NASA Center for AeroSpace Information (CASI)
Parkway Center
7121 Standard Drive
Hanover, MD 21076-1320

**FRANCE**
O.N.E.R.A. (ISP)
29, Avenue de la Division Leclerc
BP 72, 92322 Châtillon Cedex

**GRECE (Correspondant)**
Hellenic Ministry of National Defence
Defence Industry Research & Technology General Directorate
Technological R&D Directorate
D.Soutsou 40, GR-11521, Athens

**HONGRIE**
Department for Scientific Analysis
Institute of Military Technology
Ministry of Defence
H-1525 Budapest P O Box 26

**ISLANDE**
Director of Aviation
c/o Flugrad
Reykjavik

**ITALIE**
Centro documentazione tecnico-scientifica della Difesa
Via Marsala 104
00185 Roma

**LUXEMBOURG**
*Voir* Belgique

**NORVEGE**
Norwegian Defence Research Establishment
Attn: Biblioteket
P.O. Box 25, NO-2007 Kjeller

**PAYS-BAS**
NDRCC
DGM/DWOO
P.O. Box 20701
2500 ES Den Haag

**POLOGNE**
Chief of International Cooperation Division
Research & Development Department
218 Niepodleglosci Av.
00-911 Warsaw

**PORTUGAL**
Estado Maior da Força Aérea
SDFA - Centro de Documentação
Alfragide
P-2720 Amadora

**REPUBLIQUE TCHEQUE**
VTÚL a PVO Praha /
Air Force Research Institute Prague
Národní informační středisko obranného výzkumu (NISČR)
Mladoboleslavská ul., 197 06 Praha 9

**ROYAUME-UNI**
Defence Research Information Centre
Kentigern House
65 Brown Street
Glasgow G2 8EX

**TURQUIE**
Millî Savunma Başkanliği (MSB)
ARGE Dairesi Başkanliği (MSB)
06650 Bakanliklar - Ankara

## AGENCES DE VENTE

**NASA Center for AeroSpace Information (CASI)**
Parkway Center
7121 Standard Drive
Hanover, MD 21076-1320
Etats-Unis

**The British Library Document Supply Centre**
Boston Spa, Wetherby
West Yorkshire LS23 7BQ
Royaume-Uni

**Canada Institute for Scientific and Technical Information (CISTI)**
National Research Council
Document Delivery
Montreal Road, Building M-55
Ottawa K1A 0S2, Canada

Les demandes de documents RTO ou AGARD doivent comporter la dénomination "RTO" ou "AGARD" selon le cas, suivie du numéro de série (par exemple AGARD-AG-315). Des informations analogues, telles que le titre et la date de publication sont souhaitables. Des références bibliographiques complètes ainsi que des résumés des publications RTO et AGARD figurent dans les journaux suivants:

**Scientific and Technical Aerospace Reports (STAR)**
STAR peut être consulté en ligne au localisateur de ressources uniformes (URL) suivant:
http://www.sti.nasa.gov/Pubs/star/Star.html
STAR est édité par CASI dans le cadre du programme NASA d'information scientifique et technique (STI)
STI Program Office, MS 157A
NASA Langley Research Center
Hampton, Virginia 23681-0001
Etats-Unis

**Government Reports Announcements & Index (GRA&I)**
publié par le National Technical Information Service
Springfield
Virginia 2216
Etats-Unis
(accessible également en mode interactif dans la base de données bibliographiques en ligne du NTIS, et sur CD-ROM)

NORTH ATLANTIC TREATY ORGANIZATION

RESEARCH AND TECHNOLOGY ORGANIZATION
BP 25 • 7 RUE ANCELLE
F-92201 NEUILLY-SUR-SEINE CEDEX • FRANCE
Telefax 0(1)55.61.22.99 • E-mail mailbox@rta.nato.int

**DISTRIBUTION OF UNCLASSIFIED RTO PUBLICATIONS**

NATO's Research and Technology Organization (RTO) holds limited quantities of some of its recent publications and those of the former AGARD (Advisory Group for Aerospace Research & Development of NATO), and these may be available for purchase in hard copy form. For more information, write or send a telefax to the address given above. **Please do not telephone.**

Further copies are sometimes available from the National Distribution Centres listed below. If you wish to receive all RTO publications, or just those relating to one or more specific RTO Panels, they may be willing to include you (or your organisation) in their distribution.

RTO and AGARD publications may be purchased from the Sales Agencies listed below, in photocopy or microfiche form. Original copies of some publications may be available from CASI.

## NATIONAL DISTRIBUTION CENTRES

**BELGIUM**
Coordinateur RTO - VSL/RTO
Etat-Major de la Force Aérienne
Quartier Reine Elisabeth
Rue d'Evère, B-1140 Bruxelles

**CANADA**
Director Research & Development Communications & Information Management - DRDCIM 3
Dept of National Defence
Ottawa, Ontario K1A 0K2

**CZECH REPUBLIC**
VTÚL a PVO Praha / Air Force Research Institute Prague
Národní informační středisko obranného výzkumu (NISČR)
Mladoboleslavská ul., 197 06 Praha 9

**DENMARK**
Danish Defence Research Establishment
Ryvangs Allé 1, P.O. Box 2715
DK-2100 Copenhagen Ø

**FRANCE**
O.N.E.R.A. (ISP)
29 Avenue de la Division Leclerc
BP 72, 92322 Châtillon Cedex

**GERMANY**
Streitkräfteamt / Abteilung III
Fachinformationszentrum der Bunderswehr, (FIZBw)
Friedrich-Ebert-Allee 34
D-53113 Bonn

**GREECE (Point of Contact)**
Hellenic Ministry of National Defence
Defence Industry Research & Technology General Directorate
Technological R&D Directorate
D.Soutsou 40, GR-11521, Athens

**HUNGARY**
Department for Scientific Analysis
Institute of Military Technology
Ministry of Defence
H-1525 Budapest P O Box 26

**ICELAND**
Director of Aviation
c/o Flugrad
Reykjavik

**ITALY**
Centro documentazione tecnico-scientifica della Difesa
Via Marsala 104
00185 Roma

**LUXEMBOURG**
*See* Belgium

**NETHERLANDS**
NDRCC
DGM/DWOO
P.O. Box 20701
2500 ES Den Haag

**NORWAY**
Norwegian Defence Research Establishment
Attn: Biblioteket
P.O. Box 25, NO-2007 Kjeller

**POLAND**
Chief of International Cooperation Division
Research & Development Department
218 Niepodleglosci Av.
00-911 Warsaw

**PORTUGAL**
Estado Maior da Força Aérea
SDFA - Centro de Documentação
Alfragide
P-2720 Amadora

**SPAIN**
INTA (RTO/AGARD Publications)
Carretera de Torrejón a Ajalvir, Pk.4
28850 Torrejón de Ardoz - Madrid

**TURKEY**
Millî Savunma Başkanliği (MSB)
ARGE Dairesi Başkanliği (MSB)
06650 Bakanliklar - Ankara

**UNITED KINGDOM**
Defence Research Information Centre
Kentigern House
65 Brown Street
Glasgow G2 8EX

**UNITED STATES**
NASA Center for AeroSpace Information (CASI)
Parkway Center
7121 Standard Drive
Hanover, MD 21076-1320

## SALES AGENCIES

**NASA Center for AeroSpace Information (CASI)**
Parkway Center
7121 Standard Drive
Hanover, MD 21076-1320
United States

**The British Library Document Supply Centre**
Boston Spa, Wetherby
West Yorkshire LS23 7BQ
United Kingdom

**Canada Institute for Scientific and Technical Information (CISTI)**
National Research Council
Document Delivery
Montreal Road, Building M-55
Ottawa K1A 0S2, Canada

Requests for RTO or AGARD documents should include the word 'RTO' or 'AGARD', as appropriate, followed by the serial number (for example AGARD-AG-315). Collateral information such as title and publication date is desirable. Full bibliographical references and abstracts of RTO and AGARD publications are given in the following journals:

**Scientific and Technical Aerospace Reports (STAR)**
STAR is available on-line at the following uniform resource locator:
http://www.sti.nasa.gov/Pubs/star/Star.html
STAR is published by CASI for the NASA Scientific and Technical Information (STI) Program
STI Program Office, MS 157A
NASA Langley Research Center
Hampton, Virginia 23681-0001
United States

**Government Reports Announcements & Index (GRA&I)**
published by the National Technical Information Service
Springfield
Virginia 22161
United States
(also available online in the NTIS Bibliographic Database or on CD-ROM)

*Printed by Canada Communication Group Inc.*
*(A St. Joseph Corporation Company)*
*45 Sacré-Cœur Blvd., Hull (Québec), Canada K1A 0S7*